DEPARTMENT OF THE AIR FORCE
HEADQUARTERS, 88TH AIR BASE WING (AFMC)
WRIGHT-PATTERSON AIR FORCE BASE, OHIO

26 August 2014

88 CS/SCOKIF
3810 Communications Blvd
Wright-Patterson AFB OH 45433-5767

Mr. John Greenwald
[redacted]

Dear Mr. Greenewald,

This is in response to your 30 May 2014 Freedom of Information Act (FOIA) request for copy of "ADO358634 detailed Test Objectives Project Able – 3 Earth, Satellite Report dated 7 July 1959 Report number TR-59-0000-00556". The FOIA control number assigned to this request is 2014-03838-F ST1.

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DETAILED TEST OBJECTIVES

PROJECT ABLE-3

Missile 134

Revised 7 July 1959

SPACE TECHNOLOGY LABORATORIES, INC.
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DETAILED TEST OBJECTIVES

PROJECT ABLE-3, EARTH SATELLITE

Revised 7 July 1959

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ABSTRACT

The Detailed Test Objectives and flight test plans for Project Able-3, earth satellite vehicle for testing the Able-4 deep space probe configuration and gathering scientific information of propagation experiments and space environment, are contained within this document. Project Able-3 will be conducted as a part of the program directed by the National Aeronautics and Space Administration (NASA) and is a joint project representing the contributions of a number of cooperating agencies.

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1.0 TEST PLAN

During early August 1959, an Able-3 test vehicle will be launched from Stand 17A at CCMTA along a trajectory that will place an instrumented payload into an elliptical orbit varying from 145 to 19,500 nautical miles above the earth. This flight will also demonstrate the satisfactory performance of the Able-4 vehicle configuration which is very nearly identical to Able-3. The launch of the Able-3 should be between 0900 and 1100 EST so that there will be no eclipse of the satellite for at least 30 days following launch. The selected firing azimuth of 48° will establish a satellite orbit with a high inclination angle, thereby providing radiation information and other data in the northern latitudes. A high apogee is desired for better data coverage. The trajectory has been designed so that a satisfactory orbit will be obtained without ignition of the fourth-stage engine. Payload instrumentation will gather data of space environment encountered by the satellite and telemeter these data to ground tracking stations. Various propagation experiments will also be conducted using the satellite as a transmitting station.

WS 315A Missile 134 is being modified for use on the four-stage Able-3 vehicle. The test vehicle will not utilize a steering guidance system; the first and second stages are controlled by a programmed autopilot and the third and fourth stage by spin stabilization. The four stages of the test vehicle are as follows:

a. Stage I

The first stage will be the aforcmentioned WS 315A (Thor) missile, less the AC Spark Plug inertial guidance system, with a modified autopilot control system. This stage will boost the vehicle to an inertial velocity of approximately 16,000 ft/sec. Flight readiness firing of the first-stage engine is not planned.

b. Stage II

The second stage will be essentially the same as that used on projects Able and Able-1, less the accelerometer control. It consists of an Aerojet engine, autopilot control, instrumentation, and spin rockets, and will boost the vehicle to an inertial velocity of 24,030 ft/sec. An STL space guidance system (SGS) transponder will be carried (open loop) to test its functioning prior to its use on the Able-4 launching.

c. Stage III

The third stage will be the same as that used on Project Able-1, i.e., an Allegheny Ballistic Laboratory 248 A4 solid-propellant rocket engine. At third-stage burnout, the payload will have reached a velocity of approximately 33,368 ft/sec which is sufficient to enable the payload to attain an apogee of 19,585 nautical miles above the surface of the earth.

d. Stage IV

The fourth stage utilizes a new design which will enable the use of solar cells in the power supply system. The stage contains an Atlantic Research Corporation solid-propellant injection rocket, payload electronics, and solar cell power supply system. The injection rocket engine will be ignited by command under one of several specified conditions. The payload carries a doppler and command receiver-transmitter system which will be tracked by the Advanced Guidance Studies (AGS) ground station at AFMTC and by the ground stations at Hawaii and Manchester. This will permit range and range rate measurements from these stations after the payload is in orbit. Figure 1 is an artist's conception of the payload in orbit above the earth.

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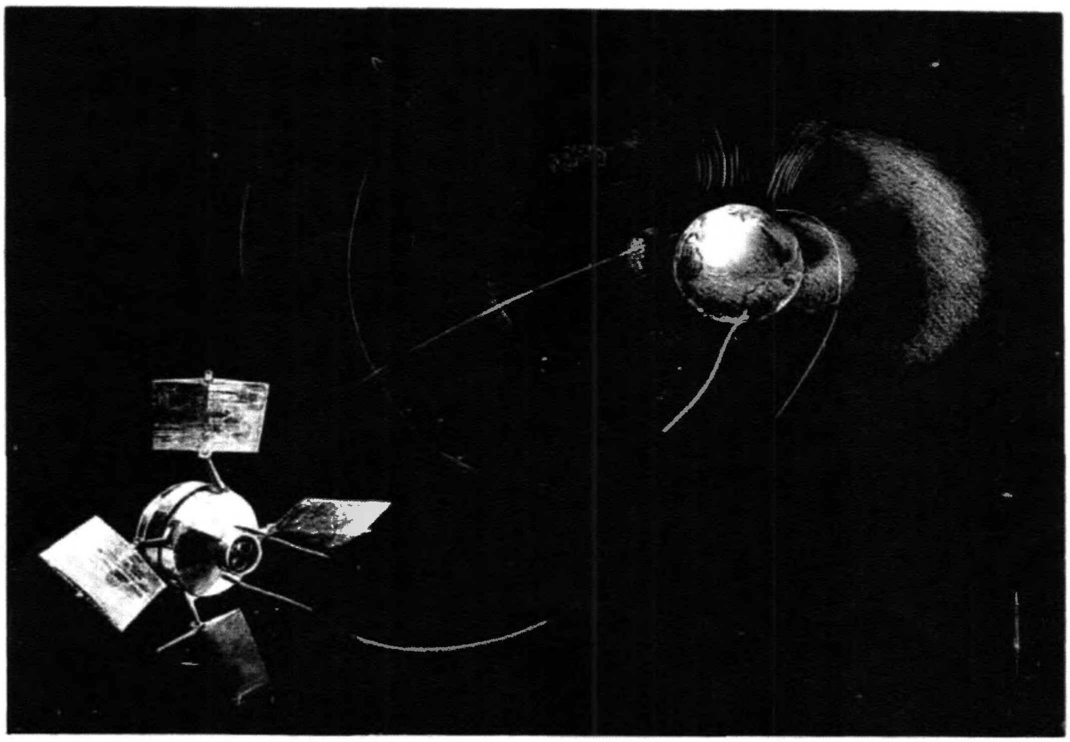

Figure 1. Artist's Conception of the Able-3 Payload Passing Through the Radiation Belts.

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2.0 TEST OBJECTIVES

2.1 Primary Test Objectives

2.1.1 Place an instrumented payload into an elliptical orbit about the earth to make scientific measurements of the environment encountered at altitudes from approximately 500 to 20,000 nautical miles.

Data Required

Telemetry transmissions of temperatures, micrometeorite flux and momentum, magnetic field, radiation flux and species, mapping of whistler mode propagation, and position of the satellite relative to the earth by means of fascimile.

2.1.2 Demonstrate satisfactory operation of the test vehicle, payload, and supporting ground stations, to be utilized for two Able-4 launchings scheduled for later during 1959. This includes the following:

a. Demonstrate satisfactory operation of the payload equipment to be used on the Able-4 launchings. This consists of the [redacted] transmitter, digital instrumentation system, solar power supply system, doppler and command system, and the experiment sensors.

Data Required

Key measurements as indicated in Appendix B.

b. Demonstrate satisfactory open-loop operation of the second-stage guidance transponder in conjunction with the Advanced Guidance Studies ground station.

Data Required

Second-stage telemetry data as indicated in Appendix B.

c. Demonstrate satisfactory performance of the third-stage propulsion system.

Data Required

Telemetry data from the second- and fourth-stage telemetry systems and tracking data derived from the doppler and command system

d. Demonstrate satisfactory operation of the special tracking stations and equipment which will be used to support the Able-4 launchings.

Data Required

Tracking and telemetry data from the various ground stations.

e. Demonstrate satisfactory operation of the second/third- and third/fourth-stage separation mechanisms

Data Required

Telemetered event data as indicated in Appendix B.

f. Demonstrate satisfactory operation of the second-stage autopilot control system and the spin control system.

Data Required

Second-stage telemetry data as indicated in Appendix B.

2 2 Secondary Test Objectives

2.2.1 Conduct electromagnetic propagation experiments from an earth satellite to determine the propagation characteristics of the ionosphere and troposphere.

Data Required

a. Electron density at the satellite, as determined from ground station reception of the doppler frequency shift from two coherent transmitters in the payload.

b. Measurement of the faraday rotation (caused by a change in the total ions along the propagation path from payload to ground).

c. Amplitude and phase fluctuations induced in the ionosphere, as measured by reception of payload transmissions at the National Bureau of Standards Laboratory in Boulder, Colorado.

2.2.2 Evaluate operation of portions of the Able-3 test vehicle which have operated reliably on previous programs; also to evaluate Able-3 equipment which will not be carried on the Able-4 vehicle. These secondary objectives will include the following:

a. Evaluate performance of the second-stage propulsion system.

Data Required

Key measurements as indicated in Appendix B.

b. Demonstrate satisfactory performance of the fourth-stage injection rocket.*

Data Required

Fourth-stage telemetry data as indicated in Appendix B.

c. Evaluate operation of the first/second-stage separation mechanism.

Data Required

Telemetered event data as indicated in Appendix B, and ground tracking data.

2.2.3 Evaluate first-stage airframe, propulsion, and modified autopilot control systems.

Data Required

WS 315A Thor measurements as indicated in Appendix B.

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*Note: The rocket will be ignited only under one of the following conditions:

(1) If burnout conditions are such that a one-week satellite lifetime cannot be assured, the injection rocket will be fired.

(2) If the rocket is not fired during the launch phase and a six-month satellite lifetime cannot be assured, the injection rocket will be fired at a proper time in one of the subsequent orbits.

(3) If the satellite life expectancy is greater than six months, the injection rocket will be fired only after agreement has been reached between AFBMD/NASA/STL.

3.0 ABLE-3 EXPERIMENTS

Essentially two classes of experiments will be conducted with the Able-3 payload. These are: (1) environmental experiments in which data of the space environment encountered by the satellite will be gathered and telemetered to ground stations; and (2) propagation experiments in which two of the payload transmitters will be used to transmit signals to Hawaii and possibly Boulder, Colorado. Payload transmitter operation will be controlled and the payload functions selected by use of the doppler and command system. Descriptions of the experiments are as follows:

3.1 Space Environment Experiments

Experiments to measure and telemeter the space environment encountered by the payload will consist of the following:

a. Micrometeorite Flux and Momentum

An apparatus similar to that used on Able-1 (flights 1 and 2) will be utilized to count impacts of micrometeorites above about 10^{-4} gr cm/sec momentum. Two momentum levels will be measured. Over-all weight of the equipment is 0.9 pound. The average power requirement is 70 mw.

b. Magnetometer (Search-Coil)

An STL search-coil magnetometer will be used in conjunction with a flux-gate magnetometer (paragraph c, below) to enable mapping of the vector magnetic field. Continuous measurements will be made of the magnetic field and its direction. The weight of this equipment is 1.1 pound; power consumption is 50 mw.

c. Magnetometer (Flux-Gate)

This equipment will be used in conjunction with the STL search-coil magnetometer to measure the spin axis component of the magnetic field. Its weight is 2.2 pounds; its power consumption is 120 mw.

d. Vehicle Position Determination

A facsimile system consisting of both optical and electronic equipment will be contained in the payload to determine the position of the vehicle relative to the earth. This equipment is a modification of the facsimile system used in the Able-1 lunar probe. Transmitted pictures of the earth will have a resolution of approximately 5 miles and will provide meteorological information (such as cloud cover). Equipment weight is 1.9 pounds; power consumption is 210 mw.

e. Ion Chamber and Geiger Tube

An ionization chamber developed by the University of Minnesota will be carried to measure the total radiation flux. In conjunction with this chamber, the University of Minnesota is also supplying a Geiger-Muller tube for count rate. The combination of these two instruments will furnish mean specific ionization per particle. Weight of this equipment is 2.1 pounds; power consumption is 140 mw.

f. Scintillation Counter

An STL scintillation counter will be used to determine the total radiation flux encountered. Shielding used on this experiment will be of a different material than that used on the University of Minnesota experiment. Equipment weight is 2.5 pounds; power required is 320 mw.

g. Cosmic Ray

A triple-coincidence proportional counter telescope, designed by the University of Chicago, will be used to obtain a total count of charged particles above two energy thresholds. Except for a possible change in shielding, this is the same experiment carried on Able-1. The seven counters are arranged concentrically to provide single-incidence and triple-coincidence measurements. Weight is 5.3 pounds, and the power requirement is 190 mw.

h. Aspect Indicator

This equipment is a phase comparator which measures the phase relationship between the output of a photoelectric diode "sun scanner" and the search-coil magnetometer; this will provide the "H" direction of the magnetic field encountered. Weight of the equipment is 0.7 pound; power required is 24 mw.

i. VLF Propagation

A VLF receiver furnished by Stanford University will be utilized to [redacted] This will enable study of the dispersive properties of the atmosphere at very low frequencies. Equipment weight is 0.5 pound and power estimate is 86 mw.

j. Other Measurements

Temperature readings of the payload compartment and of the paddles which contain the solar cells will be telemetered from the satellite together with performance measurements on the solar cell and battery system.

3.2 Propagation Experiments

The propagation experiments will utilize one-way transmission from the Able-3 payload transmitters to ground receiving stations. Three types of propagation measurements will be made as follows:

a. Electron Density

[redacted] passes through space where there are no electrons, the doppler shift is exactly proportional to frequency. The presence of electrons, however, forms a "dielectric" medium. This dielectric will have a larger effect at low frequencies than it will at higher frequencies. Careful comparison of the doppler effect at two widely separated frequencies will thereby provide a measure of ion density.

b. Faraday Rotation

The faraday rotation caused by a change in the total ions along the propagation path from payload to ground is to be measured in Hawaii. This measurement involves observation of the rotation of the plane of polarization [redacted] This rotation will be relatively slow and will be recorded automatically.

c. Amplitude and Phase Fluctuation

A third propagation experiment which may be made will use signals [redacted] Two receivers spaced on a 475-meter base line will be used to measure amplitude and phase fluctuations induced by the ionosphere. Ground equipment for this experiment is at the National Bureau of Standards Laboratory in Boulder, Colorado.

4.0 FLIGHT PLAN

A description of the trajectory-shaping parameters and staging events during powered and free-flight of the Able-3 test vehicle is given below:

Time (seconds)	Action	Equipment Initiating Action
X + 0	First motion of test vehicle; launched in vertical direction.	
X + 2 to X + 9	During vertical rise, vehicle effects a programmed roll maneuver of approximately 63° to a launch azimuth of 48° true.	DAC Programmer
X + 10	Vehicle programmed into an approximation of a gravity turn (by use of a 4-stage pitch program).	DAC Programmer
X + 140	6.7 ± 0.1g accelerometer actuated for arming of second-stage engine start; 46 ± 4-second timer started.	Missile Acceleration of 6.7g + 0.8 ± 0.3 second
X + 140	Stage I pitch program completed; missile begins flying a constant-attitude trajectory.	DAC Programmer
X + 160	Stage I main engine cutoff. First/second-stage sequence started which results in blowing blast doors and starting a 2-second time delay.	90% chamber pressure switch and associated relays. (Occurs upon incipient depletion of propellant)
X + 160 to X + 162	Stage I vernier engines propel vehicle and maintain a positive head on the second-stage propulsion system to assure reliable ignition.	
X + 162	Stage II engine start signal, uncage Stage II gyros, start 100-second timer, and start 0.3°/second pitch-down program.	2-second time delay relay
X + 162 +	The first/second stage separation bolts are blown and a 10-second timer is started; and Stage II pitch down program of 0.40°/second is started.	Chamber pressure switch set to operate at 60% of Stage II engine thrust. (TPS with TVS_2 backup)

Time (seconds)	Action	Equipment Initiating Action
X + 162.3	Separation of Stages I and II completed. The predicted impact range for the first stage is approximately 1280 nautical miles downrange	
X + 172	Pyrotechnic in the helium tanks ignited and cutoff enabled.	10-second timer
X + 187	Nose fairing, covering the third and fourth stages, jettisoned.	46-second timer
	The 4-minute timer in Stage III started.	Lanyard attached to nose fairing
X + 262	Stop pitch program	100-second timer
X + 275.8	Burnout of Stage II engine occurs; fires cable cutters, permitting solar paddle erection. A 2-second and 3.1-second timer are started. SGS transmitter turned off.	Stage II propellant depletion (TPS)
X + 276.9 +	Solar cell paddles extended and locked; payload transmitter activated.	Springs and micro-switch
X + 277.8	Spin rockets on Stage II ignited to enable spin stabilization of the third and fourth stages, and second-stage autopilot gyros caged.	2-second timer
X + 278.5	Ignition of the Stage III rocket motor occurs; simultaneously, the blowing of Stages II and III explosive bolts and nozzle shroud occurs. (Predicted impact range for the second stage is 7153 nautical miles downrange; however, it is expected to disintegrate upon re-entering the atmosphere).	3.1-second timer
X + 315.7	Burnout of Stage III rocket motor occurs; vehicle has reached an inertial velocity of 33,368 ft/sec.	Depletion of propellant
X + 427	Separation bolt blows, detaches Stage III from Stage IV.	4-minute timer started at X + 187 and separation spring

Time (seconds)	Action	Equipment Initiating Action
	Arm Stage IV ordnance	Switch actuated by physical separation
X + 427 +	Stage IV continues on a spin-stabilized constant-attitude trajectory	
*See Note	Injection rocket ignition	By command from Manchester

*Note: This event will occur in accordance with the conditions specified in the note on Page 6.

5.0 TEST VEHICLE CONFIGURATION

The Able-3 test vehicle is a four-stage missile, three stages of which are used to impart sufficient velocity to the payload (and fourth stage) to place it into an elliptical orbit about the earth. The fourth-stage injection rocket may then be utilized to increase the perigee of the orbit and to give the payload, now an earth satellite, a longer life expectancy. An outline drawing of the test vehicle with specific performance parameters is shown in Figure 2. The configuration of the four stages is as follows:

a. Stage I

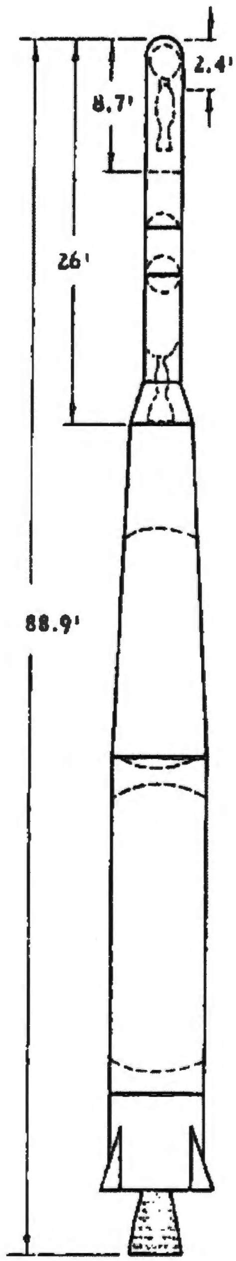
The first stage is a standard Thor missile airframe, propulsion, and control system without an inertial guidance system. Guidance will be accomplished by use of roll and pitch programmers. The control system has been modified to accommodate the dynamics of the four-stage vehicle by relocation of the rate gyros and redesign of the compensation networks.

The NAA liquid-propellant rocket engine will cut off on incipient exhaustion of either propellant, i.e., when main engine chamber pressure drops to 90 per cent of nominal.

b. Stage II

The second stage is similar to that used on the Project Able and Able-1 launchings; however, engine cutoff will occur at incipient fuel exhaustion. The second stage is comprised of two main sections: a propulsion section and a guidance section. Propulsion is provided by a model AJ10-101A Aerojet liquid-propellant system. Inhibited white fuming nitric acid (IWFNA) and unsymmetrical dimethyl hydrazine (UDMH), which are hypergolic, are injected into the second-stage thrust chamber by a helium pressure system. Pitch and yaw control of the vehicle is achieved by gimballing the second-stage engine thrust chamber; roll control is achieved by "on-off" discharge high-pressure helium through roll-control nozzles. A new lightweight programmed autopilot is used for guidance.

Eight Atlantic Research Corporation .5KS 130 spin rockets are incorporated on the second stage. These are ignited two seconds after



FOURTH STAGE

ARC 1KS 420 Injection Rocket:

Thrust at Altitude	434 lb
Vacuum I_{sp}	248 lb-sec/lb
Total Impulse	498 lb-sec
Burnout Acceleration	3.3 g

THIRD STAGE

ABL-248 A4 Engine:

Thrust at Altitude	3150 lb
Vacuum I_{sp}	250.5 lb-sec/lb
Total Impulse	116,400 lb-sec
Burnout Acceleration	16.3 g

SECOND STAGE

AGC 10-101A Engine:

Thrust (at altitude)	7670 lb
Thrust (mass burnout)	7327 lb
Vacuum I_{sp} (nom.)	271 lb-sec/lb
Total Impulse (nom.)	8.69×10^5 lb/sec
Burnout Acceleration	4.48 g

FIRST STAGE - THOR

Rocketdyne MB-1 Engine:

Sea Level Thrust	153,248 lb
Sea Level I_{sp} (nom.)	248.7 lb-sec/lb
Vacuum I_{sp} (nom.)	289.7 lb-sec/lb
Total Impulse (nom.)	2.67×10^7 lb-sec
Burnout Acceleration	12.5 g

Figure 2. Outline and Performance Parameters of Able-3 Test Vehicle.

second-stage engine cutoff, and impart a 2.8-rps spin to the second, third, and fourth stages to achieve spin stabilization of the third and fourth stages. These rockets have been relocated nearer the center of gravity of the joined second, third, and fourth stages than those used on the Able-1 vehicles.

A guidance transponder will be carried on the second stage. It will be operated on an open-loop basis in conjunction with the Advanced Guidance Studies ground station at Cape Canaveral. Simulated guidance system steering commands will be transmitted from the ground during powered flight and the

[redacted]

c. Stage III

The third stage uses an Allegheny Ballistic Laboratory 248-A4 solid-propellant rocket identical to that used on Able-1. Nominal burning time is approximately 38 seconds; burnout occurs upon propellant depletion. Control is obtained by spin stabilization from the second-stage spin rockets. Both the third and fourth stages are covered by an aerodynamic nose fairing which is jettisoned after first-stage separation.

d. Stage IV

The fourth stage (and payload) or the Able-3 test vehicle consists of essentially the following:

Structure
Propulsion System
Solar Cell and Battery System
Electronics

(1) Structure

The structure consists of a 29-inch diameter approximate sphere, flanked by four paddles equally spaced around the sphere's equator. The paddles will consist of a light aluminum spar into which are fastened modular pallets of cells. The base of the pallet is aluminum honeycomb to which solar cells are fastened on both sides. The paddles are canted to maximize the average number of cells normal to the sun's radiation. The sphere is composed of a central platform, a support structure, a lower cover, and an upper cover. The central platform will be made from a fiberglas honeycomb. The structure consists of a welded aluminum tube truss with two circumferential rings to transfer load from the platform and paddle hinge brackets to the truss. A ring is provided at the equator of the sphere for attachment of the upper and lower covers (which are thin-formed sheet metal). All electronic equipment and the storage batteries will attach to the central platform. The four paddles will furnish a larger surface area for the solar cells and an increased roll moment of inertia in lieu of a larger diameter sphere. They will also enable better temperature control of the sphere, since the surface of the sphere can be utilized to minimize the temperature variation caused by a changing solar look-angle throughout the satellite's life. The paddles are hinged so as to remain in either of two stable positions with respect to the central sphere. These positions are:

(a) Folded down symmetrically inside the aerodynamic fairing along the third-stage rocket.

(b) Extended to lie 22.5° from the sphere equatorial plane (2 up and 2 down). At Stage II shutdown the solar cell paddle cable is cut and springs will cause the paddles to move to an extended position and latch. The extending of the solar paddles also closes a microswitch for activation of the payload transmitter.

(2) Propulsion System

An Atlantic Research Corporation 1KS420 solid-propellant rocket is carried in, and attached to, the payload support structure. This rocket is ignited by command from the payload doppler and command system, and will furnish 504.3-lb-sec impulse to the payload, if ignition of this rocket is desired.

(3) Solar Cell and Battery System

Electronic components of the payload are powered by storage batteries which are kept charged by the impingement of solar radiation on banks of solar cells in the paddles. The long life expectancy of the satellite necessitates, because of weight limitations, the use of solar cells. A total of 8000 cells will be carried, with an active cell area of 12.2 square feet. Storage batteries with a minimum of 50-watt-hour capacity are required to provide for [redacted] more power than is provided by the solar cells) and to obtain uninterrupted operation of the electronic equipment when the vehicle is in the earth's shadow. Except during periods of eclipse, a capacity of 30 watts is available from the solar cells to operate the electronic equipment or to charge the storage batteries (as shown in Figure 3). Energy from the sun will be available 90 to 95 per cent of the time; the occasional eclipses will be a maximum duration of approxi- [redacted]

(4) Electronics

Electronic equipment carried by the Able-3 payload consists of the following:

(a) Sensors for measurement of space environment,

[redacted]

(e) Analog and digital instrumentation systems, including programmer,

(f) Power converters.

Experiments to be carried by the Able-3 payload are described in Section 3.0. In addition to measurements of the space environment, information on fourth-stage operation will be telemetered as indicated in Appendix B. Three trans- [redacted]

The doppler and command system will be utilized for tracking the payload and for controlling the various functions of the payload equipment. Position and velocity will be determined by use of a transponder in the payload. [redacted] to the ground by the payload transponder will be available at the ground stations.

These will allow a range measurement of the payload in orbit. Since the life of the payload is expected to be in excess of a year, a command is available for turning off all radiation from the payload to minimize the likelihood of interference with other spectrum users. Desired functions of the payload equipment will be commanded by use of a 13-digit message. One digit is for synchronization, six for the command code, and six are complements of the command code digits. This system will permit proper addressing of the Able-3 vehicle and also the Able-4 vehicles when they are launched. As an example, the [redacted]' will be as follows:

1 0 1 0 0 0 1 1 0 1 1 1 0

(a) (b) (c)

(a) Synchronized pulse

(b) Command Code

(c) Complementary digits

The synchronized pulse will have the shape of a "1" pulse and will be the first transmitted in the sequence (see Figure 4)

Figure 4 Command System Pulse Shape.

The synchronized and "1" pulses will be transmitted by the 512-cps sub-carrier "ON" and the "0" pulses will be transmitted by the subcarrier "OFF" (see Figure 5.

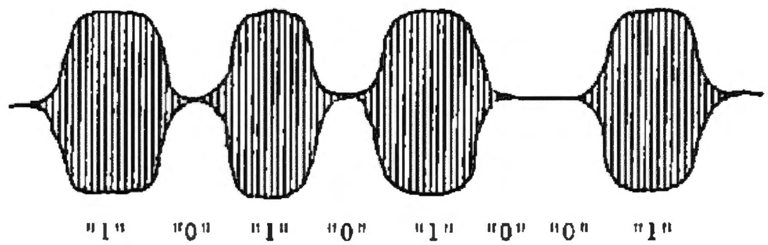


Figure 5. Command System Pulse Train.

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Command codes for the Able-3 vehicle are as follows:

COMMAND NAME	F E D C B A	CODE NO.
Digital 64 PPS	0 0 0 0 0 1	1
Digital 8 PPS	0 0 0 0 1 0	2
[redacted]		
Receiver Narrow Band	0 1 0 1 0 0	20
Clear Rocket	0 1 1 0 0 0	24
Arm Injection Rocket	0 1 1 0 0 1	25
Fire Rocket	0 1 1 1 0 0	28
[redacted]		

The first digit transmitted in the sequence is the most significant binary digit (F).

The following is a description of the operation of the previously listed commands. The command numbers given are in decimal notation and should not be confused with the octal notation used elsewhere.

1. All commands, except "Over Voltage" - Code 18 and "Fire Rocket" - Code 28, are controlled by latch-type relays on the payload.

2. "Digital 64 PPS" - Code 1, "Digital 8 PPS" - Code 2, "TV ON" - Code 3, "5-Watt Transmitter ON" - Code 9, and "Accelerometer ON" - Code 11 are all reset simultaneously by "378-mc Transmitter OFF" - Code 8.

3. Should the voltage of the solar-powered batteries decrease to 14.8 volts,
[redacted]

5. "Over Voltage" - Code 18 - applies a step pulse to a ledex stepping switch which changes the charging rate of the solar batteries.

[redacted]

6.0 FLIGHT TABLES

1. Initial Conditions

A. Launch coordinates, launch pad 17A

Latitude (geodetic)	= 28.44687°N
Longitude	= 80.56517°W
Elevation	= 0.0 feet at MSL

B. Doppler site radar coordinates (AGS station)

Latitude (geodetic) [redacted]

Longitude

C. Launch azimuth (Pad 17A)

D. Lift-off weight	= 112,257
E. Mixture ratio	= 2.212:1

F. Propellant utilization (residual usable propellant)

First stage	= 0.5 per cent**
G Thrust (sea level)	= 153,248 lb**
H Liquid oxygen regulator reference setting (sea level)	= 489 ±5 psig
I. Specific impulse (sea level)	= 250.1 sec tag value (corrected)
	= 248.7**

2. Autopilot Settings

A. Pitch program* [redacted]

	Time (sec)
1. First stage	0-10
	10-28
	28-70
	70-98
	98-140
	140-160.0

*90°T used in determining roll program.

**Values used in trajectory calculations.

	Time (sec)	rad/sec	deg/sec
2. Vernier stage	160.0-162.0	0.0	0.0
3. Second stage	[redacted]		
4. After second stage	275.8-315.7	0.0	0.0

B. Roll program

1. First stage

Azimuth required = 48.00°

Roll required = 42.00°

Roll rate

Time (sec)	
0-2	[redacted]
2-9	
9-burnout	

Note: Positive roll command is a roll clockwise as viewed from the rear of the missile during its vertical rise.

2. After first stage, no roll program

C. Yaw program, none

3. Engine Cutoff, Staging and Timer Information

A. First-stage main engine is to be cut off when the thrust chamber pressure decays to 90 per cent of nominal.

B. Initiation of staging events is to accur 2.0 ± 0.2 seconds after receipt of the signal to cut off the first-stage main engine.

C. Second-stage cutoff signal is to be given by the second-stage engine thrust pressure switch when the thrust chamber decays to 108 psi.

D. Pyrotechnic in helium tanks is to be ignited 10 ± 1 second after first/second-stage separation.

E. Nose fairing is to be jettisoned 46 ± 4 seconds after the missile has reached 6.7 ± 0.1g acceleration plus 0.8 ± 0.3 second.

F. Spin rockets are to be ignited and second-stage gyros caged at 2 ± 0.2 seconds after second-stage burnout (TPS).

G. Stage III rocket is to be ignited and Stages II/III separated at 3.1 ± 0.2 seconds after second-stage burnout (TPS).

H. Stages III/IV separation bolts are to be blown 4 minutes ± 20 seconds after the nose fairing is jettisoned.

I. Stage II pitch program is to be stopped 100 seconds after second-stage engine start signal.

7.0 MISSILE-BORNE INSTRUMENTATION

7.1 Stage I

Instrumentation equipment carried on the first stage consists of one PDM/FM telemetry set and an Azusa transponder. The PDM/FM telemetry set, which samples 30 functions at the rate of 30 times per second, will be used to provide 25 measurements channels plus three references and a keying pair. Two channels will carry three sequencing voltage indications each; the remaining channels will measure one parameter each. The r-f carrier will [redacted] airframe measurements to enable continued evaluation of the WS 315A. The measurement list is shown in Appendix B.

The Azusa transponder will provide position and velocity data during first-stage powered flight. The Azusa system used in conjunction with the IBM 709 computer will provide instantaneous impact prediction measurements for use by range safety.

7.2 Stage II

The only instrumentation equipment carried on the second stage will consist of an FM/FM telemetry set and associated transducers including an accelerometer. No electronic tracking equipment will be carried as such; tracking data, however, can be obtained from the guidance receiver. The [redacted] antenna system will consist of a double-Yagi configuration with the antennas spaced 180° apart. The system is entirely self-contained and will utilize its own batteries for primary power. Data transmitted will consist of strategic second-stage airframe, propulsion, and control measurements, plus specific events. In addition, third-stage initiation will be telemetered. The measurement list is shown in Appendix B.

7.3 Stage III

No telemetry or tracking equipment is carried on the third-stage vehicle. Various third-stage functions, however, will be telemetered during powered flight by use of second- and fourth-stage telemetry equipment.

7.4 Stage IV

[redacted] in analog form. Information telemetered is shown in Appendix B. Other equipment on the fourth stage is described in Section 5.0.

7.5 Range Safety Equipment

A standard Thor dual-destruct system is carried on the first stage. This consists of two destruct receivers, two safety and arming units, and detonating cord strands. The detonating cord will also be installed on the second stage so that both the first and second stages can be destroyed simultaneously during first-stage powered flight. An acceleration switch which locks out second-stage ignition until 0.8 second after the vehicle has reached an acceleration of approximately 6.7 g's is also carried on the second stage.

8.0 GROUND STATION INSTRUMENTATION SUPPORT

The world-wide complex of ground stations established for the Able-1 lunar probe program has been supplied additional equipment to be used on the Able-3 and Able-4 programs. These stations will provide telemetry reception and tracking data and also will be used to conduct the propagation experiments.

The Able ground station at AFMTC will provide reception of telemetry signals on a continuing basis and will also be used for checkout of the payload telemetry during countdown.

The Manchester ground station (which uses the 250-ft radio telescope at Jodrell Bank), will provide reception of telemetry signals and also precise tracking information on the payload during launch and in orbit. It will also be equipped to send commands to the payload.

The Millstone Hill, Massachusetts, radar site of the MIT Lincoln Laboratories will utilize the 85-ft parabolic antenna. This station will provide skin-tracking of the second stage by monopulse radar during launch. The Millstone station will then switch to telemetry reception after the first few minutes of the launch phase and, when the payload is in orbit, will provide telemetry reception, angular tracking information, and one-way doppler measurement.

The Hawaii station does not participate in the launch phase. After the payload is in orbit the Hawaii station will provide angular tracking information, reception of telemetry signals, and transmission of commands. It will be the only ground station instrumented for the propagation and faraday rotation experiments.

The Singapore station does not participate in the launch phase. Once the payload is in orbit the Singapore station will provide reception of telemetry signals. The Singapore antenna beamwidth is too broad for development of effective angular tracking information.

8.1 Air Force Missile Test Center

Ground instrumentation support from AFMTC will consist of missile tracking and telemetry reception as follows:

a. Azusa tracking information during first-stage powered flight.

b. FPS-16 radar skin tracking from lift-off until the end of signal reception. It is not expected that this radar will furnish data beyond the first-stage powered flight.

c. Telemetry reception of first-stage PDM/FM and second-stage FM/FM internal measurements. A 60-ft telemetry dish is suggested.

8.2 Able-3 Ground Station

The Able-3 ground station has been located at the same site at AFMTC as the Able-1 station. It contains the following equipment:

[redacted]

d. Discriminators and Sanborn recorders to permit simultaneous real-time reduction of all remote telemetry and Sanborn recording tracks for recording signal strength and time.

e. Timing equipment which furnishes timing signals to data demodulation equipment and Sanborn recorder.

[redacted]

g. Diesel power consisting of two 50-kw Caterpillar diesel-driven generators will provide all necessary power for this station. These generators are supplied by STL. Equipment will be housed in two air-conditioned trailers.

h. The ground communications facilities that will be used during launch are identical with those required for the lunar probe program. After the launch phase has been completed, continuous teletype service (or telephone service) from the trailer to the STL operations center in Los Angeles will be required to receive steering information and to transmit telemetry data.

8.3 AGS Ground Station

An STL Advanced Guidance System ground station has been located at the GE Mod I radar complex at Cape Canaveral. During the Able-3 and Able-4 launches this station will provide launch control, tracking, and commands for both the second and fourth stages. Three receiver trailers, a transmitter trailer, and a data trailer are located at this site. Operation of this station will be in conjunction with the Able-3 ground station as shown in Figure 6.

8.4 Manchester Station

The Manchester station utilizes the 250-ft radio telescope at Jodrell Bank and a motor-driven helical array furnished by STL. This station should acquire the payload within 15 minutes after launch and will be able to provide angular tracking information and telemetry recording for the latter part of the launch phase. The station will not be equipped for the propagation experiment but will be able to transmit commands to the payload.

The following equipment will be provided at Manchester:

a. The 250-ft radio telescope at Jodrell Bank is the primary antenna at this station.

[redacted]

d. Ampex tape recorder Model FR 107 (seven tracks of 10-kc bandwidth each) for permanent recording of telemetry information and local equipment parameters.

e. Doppler extraction unit with automatic printout of time and doppler frequencies.

f. Timing equipment which furnishes timing signals to data demodulation equipment and Sanborn recorder.

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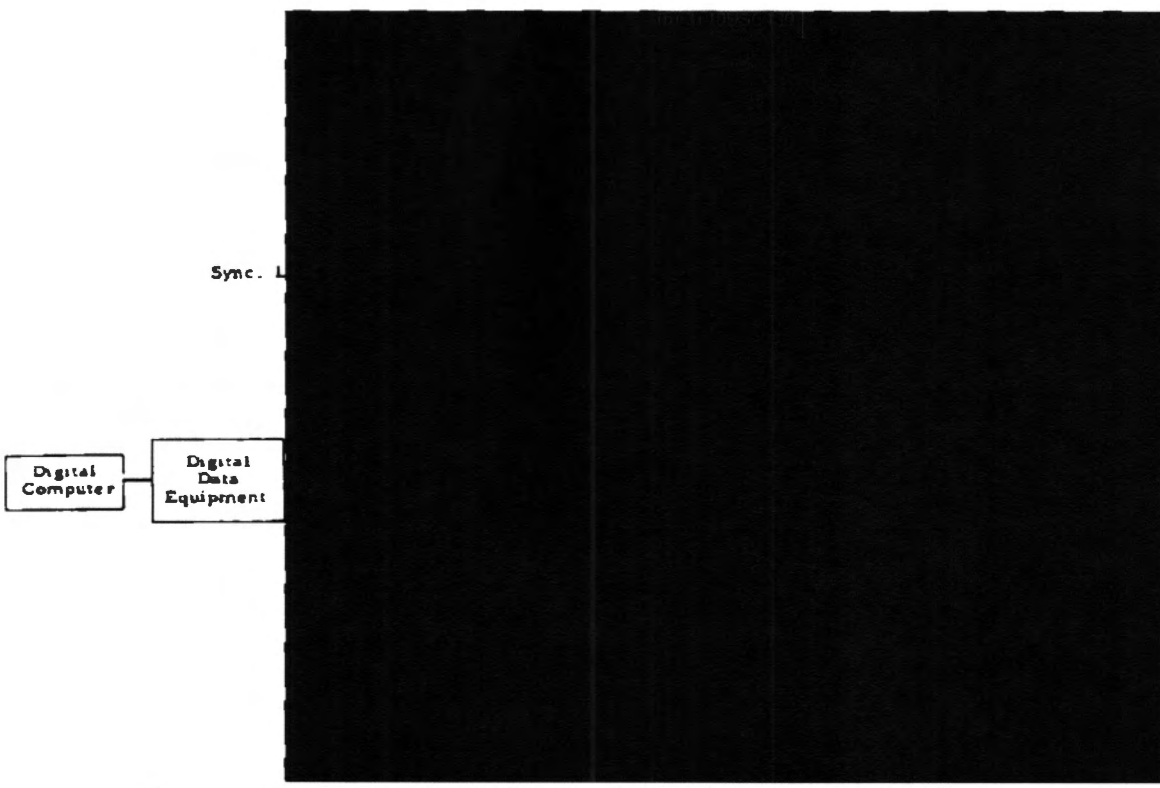

[redacted]

Figure 6. Launch Guidance and Tracking Scheme for Able-3 and Able-4.

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g. Digital data demodulation and automatic teletype tape punching for [redacted]

h. Discriminators and Sanborn recorders to permit simultaneous real-time reduction of all remote telemetry and Sanborn recording tracks for signal strength and time recording.

i. Digital command modulator with capability of generating required commands.

j. Ranging unit with 0.25-, 2- and 16-cps ranging modulation.

[redacted]

l. Primary frequency and time standard.

m. Teletype communications will be required between this station and the STL communications center in Los Angeles on a 24-hour basis so that look-angle information can be transmitted to and from STL, and so the telemetered data can be reported to STL.

n. Power for this station will be obtained from transformers and motor generator sets which were installed for the lunar probe experiments.

8.5 Millstone Station

The Millstone facility (constructed and operated by MIT) has an 85-ft parabolic antenna and monopulse radar with automatic tracking. A Microlock [redacted] added for the Able-1 (lunar probe) program.

The Millstone station should acquire the rocket within a few minutes after launch and will skin-track the second stage with the monopulse radar. The station will then switch over to reception on Microlock receivers for tracking and telemetry reception during the orbiting phase.

Facilities at the Millstone station have been expanded to permit full reception of telemetry signals. The equipment provided by STL is as follows:

a. [redacted]

b. Magnetic tape recorder for permanent recording of telemetry signals and local parameters (being supplied by MIT). Subcarrier oscillators provided by STL to permit recording of signal strength, receiver gain bias, and other local parameters.

c. Power for this station is obtained from commercial power lines. MIT laboratory facilities are used to house the equipment.

8.6 Hawaii Station

The Hawaii station was the primary station for the lunar probe program. [redacted] and analysis equipment. New equipment has been supplied to the Hawaii station. Its total capability, including both new and old equipment, is as follows:

[redacted]

b. Magnetic tape recorder for permanent recording of telemetry signals and local equipment parameters.

c. Discriminators and Sanborn recorders to permit simultaneous real-time reduction of all remote telemetry and Sanborn recording tracks for signal strength and time recording.

[redacted]

e. Doppler-measuring equipment (propagation experiment) to measure the anomaly in doppler shift caused by the payload moving through ions in space.

f. Ranging unit with 0.25-, 2- and 16-cps ranging modulation.

g. Faraday rotation is measured with special antennas, utilizing the [redacted]

h. Digital command modulator with capability of generating required commands.

i. [redacted] extraction equipment.

j. Doppler extraction unit with automatic printout of time and doppler frequencies.

k. Digital data demodulation and automatic teletype tape punching for digital information on [redacted]

l. Timing equipment which furnishes timing signals to data demodulation equipment and Sanborn recorder.

m. Primary frequency and time standard.

n. Power for station to be obtained from three existing 60-kw diesel-driven generators. (The added equipment will not require an increase in total power capability of the station for the August launch.)

o. The communications facility to STL will be a 24-hour service leased-wire teletype.

8.7 Singapore Station

The Singapore station does not have large parabolic broad-band antennas; therefore, this station will be restricted to the reception of telemetry information. The station has new equipment (in addition to Able-1) so that its total capability is:

[redacted]

c. Ampex tape recorder Model FR 107 (seven tracks of 10-kc bandwidth each) for permanent recording of telemetry information and of local equipment parameters.

d. Discriminators and Sanborn recorders to permit real-time reduction of all remote telemetry simultaneously and Sanborn recording tracks for recording signal strength and time.

e. Timing equipment which furnishes timing signals to data demodulation equipment and Sanborn recorders.

f. Digital data demodulation and automatic teletype tape punching for digital information on [redacted]

g. Power for this station is provided by two existing 60-kw diesel generators. Since this station will be on 24-hour standby, this power should be backed up by commercial power facilities at 50 cycles with a motor-generator set to transform the power to the 60 cycles 115/208 volt, 3-phase power required.

h. Ground communications will be leased radio-teletype service on a 24-hour basis. Services which were provided for the Able-1 program proved to be unreliable. An attempt is being made to replace these with trans-Pacific radio services for increased reliability.

8.8 Other Stations

Other stations expected to participate are the NASA Minitrack stations located in the launch area and the National Bureau of Standards station at Boulder, Colorado.

9.0 GROUND COMMUNICATIONS AND DATA HANDLING PROCEDURES

The BMD/STL Operations Center that was established for Project Able-1 has been expanded for use on Project Able-3. This center, now designated the Space Navigation (SpaN) Center, is located at Space Technology Laboratories, Inc., Building E, Los Angeles. Teletype and leased telephone links between this facility and the various ground stations to be utilized are shown in Figure 7.

Prior to launching, nominal trajectory data will be transmitted to these stations as well as to other cooperating stations. The data will be used to plan nominal steering periods, and will be used for antenna steering until more accurate data from actual tracking is available. After the missile is launched, the trajectory measurements from the various stations will be reduced at the site to a form suitable for transmittal to the SpaN Center. The first stations to have this data will be those stations located in and near the launching area. The data normally will be coordinated through the STL operations at AFMTC. Tracking data transmitted to the SpaN Center will be used in a continuing operation. Increasingly accurate estimates of the actual trajectory are calculated, on the IBM 704 computer of the Computer and Data Reduction Center. After more accurate trajectories are calculated, look-angle data for the various ground stations will be transmitted to these ground stations. The recalculation of trajectory and steering data will proceed on a continuing basis throughout the useful life of the Able-3 payload. The trajectory determination task will operate on a full-time basis from launch time until the time when a very accurate trajectory has been determined.

Thereafter, trajectory calculations will be made only as required to plan routine tracking operations at the ground stations. After the first few days of flight, scheduling of tracking operations probably will be established as a weekly operation. The emphasis in the initial post-launch period will generally be to quickly establish a sufficiently accurate trajectory to determine the degree of success of the operation and to insure that all tracking stations have accurate look-angle data.

In addition to trajectory data, various sites will also receive information telemetered from the test vehicle. The telemetered data will be partially reduced at the receiving site and transmitted to the SpaN Center for further interpretation.

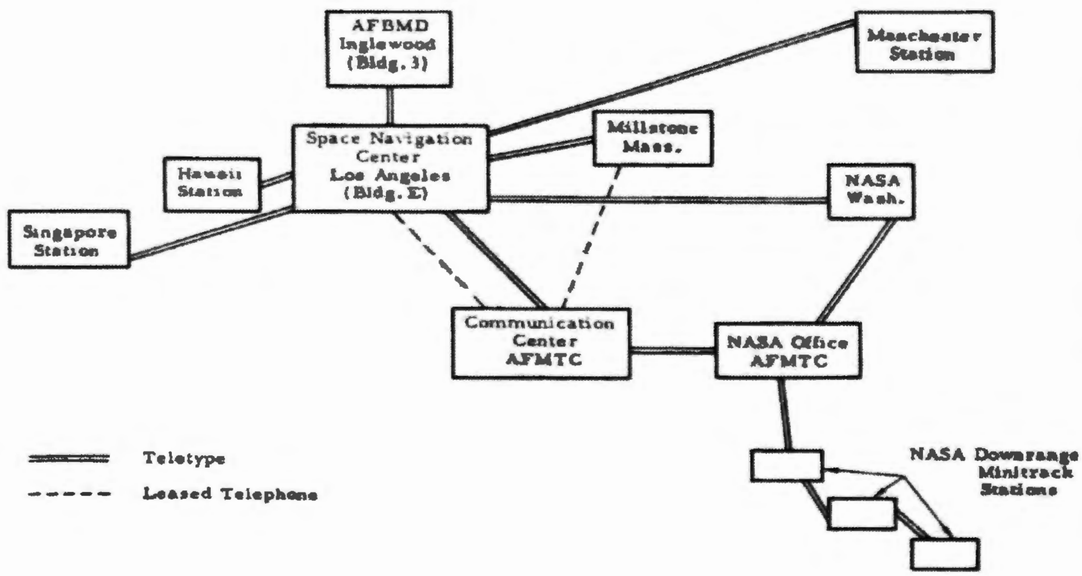

Figure 7. Project Able-3 Communication Network.
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APPENDIX A

PRELIMINARY WEIGHT BREAKDOWN ABLE-3

The following represents the latest weight estimates for the first Able-3 vehicle, using WS 315A Missile 134. Final data will be obtained from actual weight measurements in Florida prior to erection on the stand.

Stage I			
Dry Weight	7,650		
Stage II Adapter	96		
Residuals	1,426		
Nose Fairing	116.5		
Stage II Start	6.1		
Jettisoned	9,294.6	(1/2% PU as used in trajectory)	
Propellant Burned	98,126		
Gross Weight at Lift-off			107,420.6
Stage II			
Dry Weight	847.9		
Residuals	96.2		
Jettisoned Weight	944.1		
Propellant Burned	3,235.2	(2-1/2 sec bias)	
Gross Weight at Lift-off			4,179 3
Stage III			
Burned Weight	49.8		
Interstage	6.0		
Jettisoned Weight	55.8		
Propellant Burned	463.7		
Gross Weight at Lift-off			519.5
Terminal Stage			
Payload	93.7		
Structure	36.3		
Injection Rocket	5.3		
Blast Shields	0.9		
Temperature Control	1.0		
Gross Weight at Lift-off			137.2
Vehicle Gross Weight at Lift-off			112,256.6

TERMINAL STAGE WEIGHT BREAKDOWN

Over-all

Basic Electronics	52.3
Experiments	17.0
Power Source (batteries and solar cells)	32.8
Structure	27.9
Injection Rocket	5.3
Blast Shields	0.9
Temperature Control	1.0
	137.2

Basic Electronics

Transmitters, Including Antennas and Heat Sink	6.6
Receiver and Control System	9.1
Power Converter Including Heat Sinks	6.7
Telemetry System	15.1
Angular Accelerometer and Battery	3.2
Platform, Harness and Miscellaneous Hardware	11.6
	52.3

Experiments

Cosmic Ray Telescope	5.3
Search Coil and Magnetometer	1.0
Ion Chamber and Geiger Tube	2.1
Micrometeorite Momentum Spectrometer	0.9
Temperature Indicators (6)	0.1
Flux Gate Magnetometer	2.2
Scintillation Counter	2.5
Facsimile	1.9
Aspect Indicator	0.5
VLF Monitor	0.5
	17.0

Power Source

Batteries	17.8
Solar Cells	15.0
	32.8

APPENDIX B

MEASUREMENTS TO MEET DATA REQUIREMENTS FOR ABLE-3 EARTH SATELLITE WS 315A MISSILE 134

Note: Changes or additional measurements may be indicated in the Flight Test Directive as approved by the Flight Test Working Group (must measuremente will be indicated by the FTWG).

Telemetry Measurements	Test Objectives				
	Primary (2.1)		Secondary (2.2)		
First Stage (PDM/FM)	0.1	0.2	0.1	0.2	0.3
1. No. 5 Bearing Temperature					X
2. Pitch Attitude Error					X
3. Hydraulic Pressure					X
4. Pitch Rate					X
5. Pitch Main Engine Position (fine)					X
6. Vernier Roll Deflection (fine)					X
7. Sequential Voltages (No. 1)					
Fuel Float Switch Closure					X
Liquid Oxygen Float Switch Closure					X
Gas Generator Valve Closed					X
8 Sequential Voltages (No. 2)					
Main Fuel Valve Closed					X
Float Switch Arm Signal					X
Main Engine Cutoff Signal					X
9. Roll Attitude Error					X
10. Roll Rate					X
11. Turbopump Speed					X
12. Yaw Attitude Error					X
13. Gas Generator Liquid Oxygen Injector Pressure					X
14. Yaw Rate					X
15. Yaw Main Engine Position (fine)					X
16. Main Engine Chamber Pressure					X
17. Gas Generator Chamber Pressure					X
18. Vernier No. 1 Chamber Pressure					X
19. No. 5 Bearing Lube Oil Pressure					X
20. Main Liquid Oxygen Tank Pressure (top)					X
21. Main Liquid Oxygen Injector Pressure					X
22. Main Fuel Tank Pressure (top)					X
23. Absolute 5-volt Reference					X

Telemetry Measurements	Test Objectives				
	Primary (2.1)		Secondary (2.2)		
	0.1	0.2	0.1	0.2	0.3
First Stage (PDM/FM) (Continued)					
24. 400-cps Voltage (control inverter)					X
25. Fuel Pump Inlet Pressure					X
26. Main Fuel Injector Pressure					X
27. 5-volt Reference					X
28. Ground					X
Second Stage (FM/FM)					
[redacted]				X	
				X	
				X	
Accelerometer				X	
11.2 Ground					
11.3 Ground					
11.4 Integrating Accelerometer (fine shift)				X	
11.5 Angular Accelerometer (± 1.25 rad/sec^2)				X	
11.6 Zero-volt Reference				X	
11.7 Zero-volt Reference				X	
11.8 Zero-volt Reference					
11.9 Angular Accelerometer (± 1.25 rad/sec^2)				X	
11.10 Ground					
11.11 Ground					
11.12 Integrating Accelerometer (fine shift)				X	

Telemetry Measurements	Test Objectives				
	Primary (2.1)		Secondary (2.2)		
Second Stage (FM/FM) (Continued)	0.1	0.2	0.1	0.2	0.3
Channel (IRIG)					
11.13 Angular Accelerometer (±1.25 rad/sec^2)				X	
11.14 5-volt Reference					
11.15 5-volt Reference					
11.16 5-volt Reference					
[redacted] 10USC13 [redacted] (0-5 volts coded)					
a. Stage II Arm				X	
b. Nose Fairing Separate		X			
c. Pitch Up Command		X			
d Pitch Down Command		X			
e Pitch Stop Command		X			
f. Yaw Right Command		X			
g. Yaw Left Command		X			
h. Yaw Stop Command		X			
i. Spin Initiation		X			
j. Stage III Ignite, II/III Separation Bolts, Band		X			
[redacted]					
13.1 28-volt Reference (25-32 volts)					
13.2 Roll Demodulator Output ($\pm6°$)		X			
13.3 Pitch Demodulator Output ($\pm6°$)		X			
13.4 Yaw Demodulator Output ($\pm6°$)		X			
13.5 Pitch Gimbal ($\pm3°$)		X			
13.6 Yaw Gimbal ($\pm3°$)		X			
13.7 Pitch Control Field		X			
13.8 Yaw Control Field		X			
13.9 10-volts 400 Cycles Amplitude (8-12 volts)		X			
13.10 Ground				X	
13.11 Control Compartment Temperature Opposite Side (0-750°F)				X	

Telemetry Measurements	Test Objectives				
	Primary (2.1)		Secondary (2.2)		
Second Stage (FM/FM) (Continued)	0.1	0.2	0.1	0.2	0.3
Channel (IRIG)					
13.12 Control Compartment Temperature Target Side (0-750°F)				X	
13.13 Skirt Temperature Opposite Side (0-500°F)				X	
13.14 Engine Compartment Temperature Target Side (0-500°F)				X	
13.15 Skirt Temperature Target Side (0-500°F)				X	
13.16 Shroud Temperature Target Side (700-1600°F)				X	
13.17 Zero Reference					
13.18 5-volt Master Pulse					
13.19 5-volt Master Pulse					
13.20 5-volt Master Pulse					
14 (22 kc) Second Stage Events (0-5 volt coded)					
a. Lift-off				X	
b. Arm Stage II				X	
c. MECO				X	
d. Stage II Fire				X	
e. Stage II Separate Bolts				X	
f. Stage I/II Separate		X			
g. HGA, Nose Fairing		X			
h. CW and CCW Roll				X	
i. Command Enable		X			
j. Stage II Command Cutoff or TPS Shutdown				X	
k. Stage III Igniter Current		X			
l. Stage II/III Strut Release		X			

Telemetry Measurements				Test Objectives				
Fourth Stage	Transmitter			Primary (2.1)		Secondary (2.2)		
Measurement	A	B	C	0.1	0.2	0.1	0.2	0.3
Cosmic Ray (single)	D		6	X	X			
(triple)	D	2		X	X			
Micrometeorite (high momentum)	D			X	X			
Micrometeorite (low momentum)	D			X	X			
Micrometeorite			3	X	X			
Scintillation Counter	D	5		X	X			
Magnetometer (search coil)	D	1		X	X			
Univ. of Minn. (ion chamber)	D			X	X			
Univ. of Minn. (Geiger tube)	D			X	X			
Univ. of Minn.			4	X	X			
Magnetometer (flux gate)	D	3		X	X			
Magnetometer (phase)	D	4		X	X			
Aspect Indicator	D			X	X			
Accelerometer (by command only)			1	X				
Facsimile Scanner			1	X	X			
VLF (Stanford)	D	8		X	X			
Blip Strip			5	X				
a. Solar Cell Paddles Locked								
b. Stage III/IV Separation								
c. Stage IV Ignition Current								
Subcommutated Measurements (16)	D		2	X	X			
a. Paddle Temperature Outboard No. 1								
b. Paddle Temperature Outboard No. 2								
c. Paddle Temperature Inboard No. 1								
d. Shell Temperature No. 2								
e. Solar Cell Temperature								
f. Solar Cell Voltage								

Telemetry Measurements		Test Objectives				
		Primary (2.1)		Secondary (2.2)		
Fourth Stage (Continued)	Transmitter	0.1	0.2	0.1	0.2	0.3
Measurement	A B C					
g. Solar Cell Current Monitor						
h. Battery Voltage						
i. Transmitter Heat Sink Temperature						
j. Converter Receiver Sink Temperature						
k. Command Receiver Phase Error						
l. Battery Temperature						
m. Shell Temperature No. 3						
n. Shell Temperature No. 1						
o. Disk Angle						
p. Reference						

[redacted]

APPENDIX C

POWERED AND FREE FLIGHT TRAJECTORY DATA

1. Introduction

This appendix provides powered-flight and free-flight trajectory information for an Able-3 vehicle launch during August 1959. Computer print-out sheets are presented for the powered flight portion of the trajectory at the end of this appendix; various tables, graphs, and curves are presented for both the powered- and free-flight trajectories as applicable to the text.

The nominal orbit has an apogee of 19,585 nautical miles above the earth's surface, a perigee of 144 nautical miles without firing injection rocket, a period of 10 hours and 41 minutes, and an orbital inclination of 47.7°. Firing of the injection rocket would increase the perigee from 90 to 170 nautical miles, depending upon the orbital point at which the rocket is fired. The vehicle is to be launched from Stand 17A (latitude 28.44687°N, longitude 80.56517°W) at an azimuth of 48° true. This azimuth is compatible with range safety requirements and results in a satisfactory orbital inclination angle. The first- and second-stage pitch programs are shown in Section 6.0 (Flight Tables).

Table 1. Pitch Rate.

Time	[redacted]
0-10	
10-28	
28-70	
70-98	
98-140	
140-162.0	
162.0-262.0	
262.0-315.7	

This program results in [redacted] the relative velocity vector and the local radial at first-stage main engine cutoff, and an angle of attach of [redacted] This imposes no aerodynamic heating problems.

10USC

The second-stage pitch rate [redacted] at Stage III burnout. This angle represents the best compromise between apogee and perigee altitudes for a given total payload weight. A smaller second-stage pitch rate will result in a greater apogee, but will reduce the perigee while at the same time increasing the sensitivity of perigee altitude to propulsion and control system dispersions. The trajectory selected will result in a nominal perigee of 144 nautical miles in the event of the fourth-stage injection rocket not firing. This perigee will result in an orbital lifetime of over one year.

The fourth-stage injection rocket is the 1 KS 420 rocket weighing five pounds and providing an impulse of 498 pound-seconds. This results in a vectorial velocity increment of 119.7 ft/sec. This rocket would increase the perigee by 200 nautical miles if fired at first apogee. For the nominal flight the perigee without rocket firing is satisfactory. However, in the case of a malfunction, the rocket can mean the difference between no lifetime and a satisfactory life-time.

The highlights of powered-flight and free-flight trajectory are presented in Table 2. Table 3 gives the dispersions in angle, velocity, and altitude at Stage III burnout due to non-nominal engine performance, control system errors, and Stage III impulse uncertainty. Table 4 presents the dispersions in several orbital parameters due to these same disturbances. Figure 8 shows, on a Mercator projection, the nominal IIP locus through approximately 275 seconds of burning time; also indicated are the 2σ lateral deviation in the nominal IIP due to control system errors, and the 2σ elliptical areas of first- and second-stage impact. Figure 9 shows the nominal IIP locus and IIP loci for maximum rate powered turns, initiated at various times and held for 5 and 10 seconds. A stabilized turn at 90° yaw angle of attack is assumed.

Figures 10 through 25 show the variation of various trajectory parameters during powered flight. Table 5 presents IIP information for premature motor

Table 2. Selected Trajectory Information.

[redacted]

Table 2. Selected Trajectory Information (Continued).

[redacted]

Table 3. Stage III Burnout Dispersions.

[redacted]

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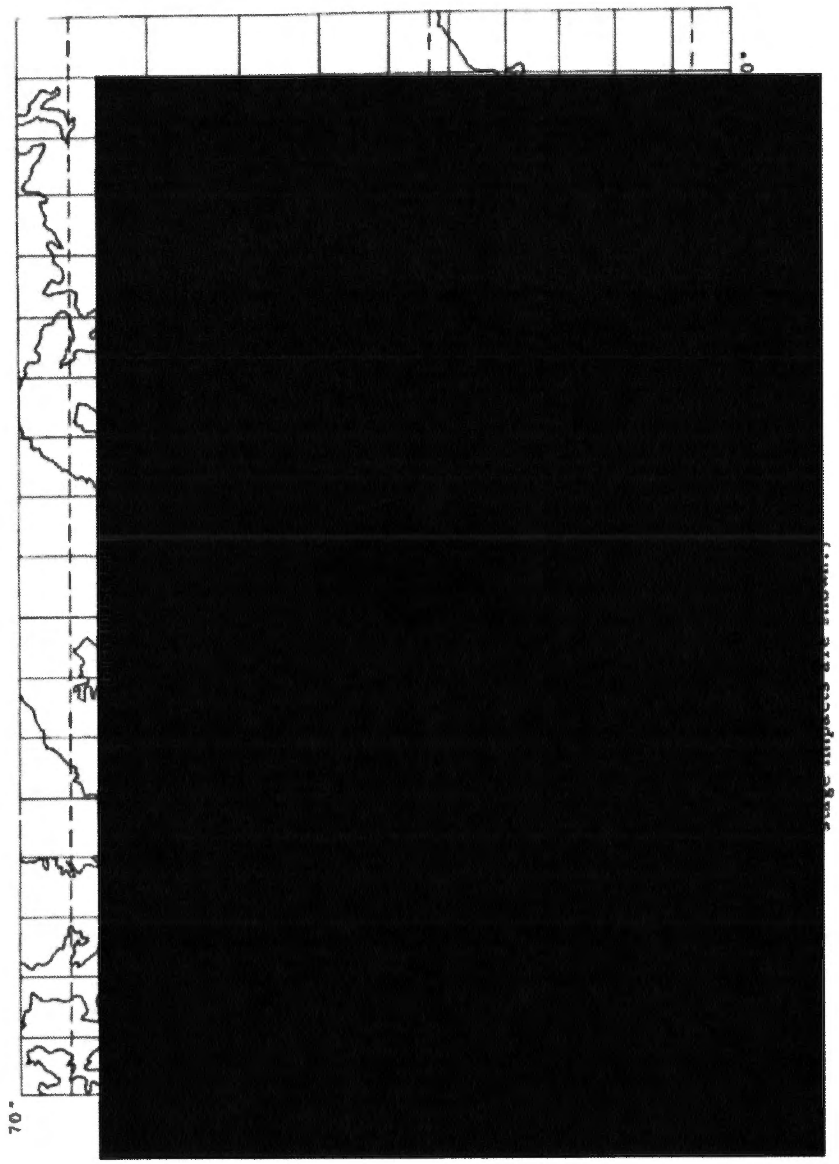
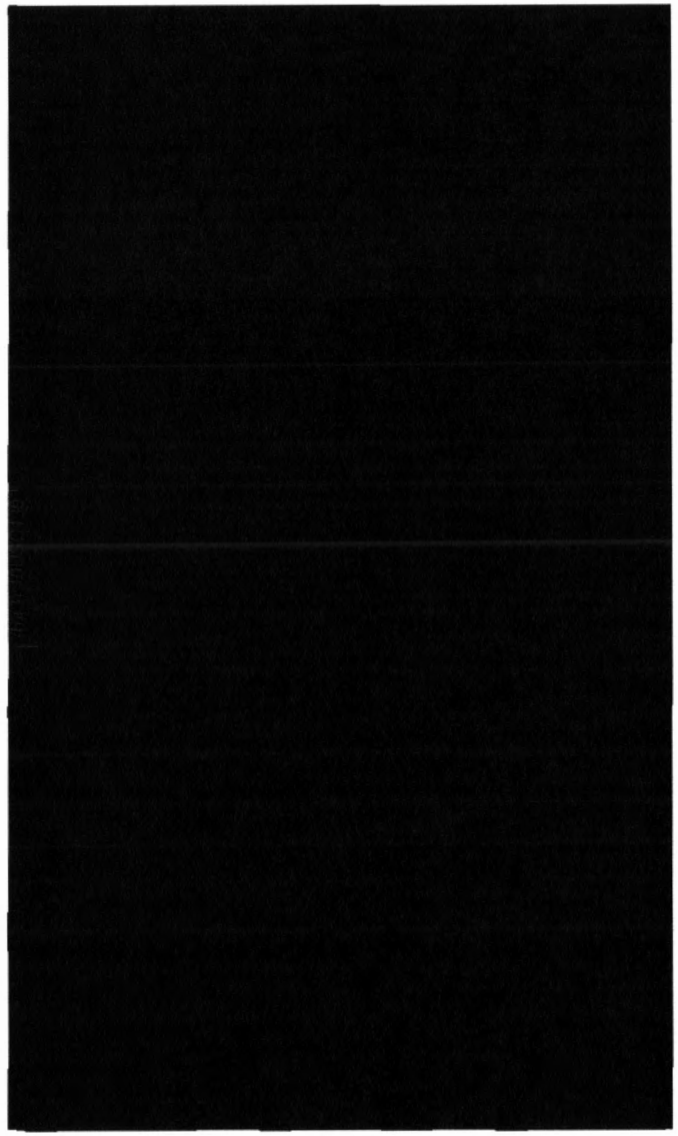
Table 4. Orbital Parameter Dispersions.

[redacted]



[redacted]

(1038)



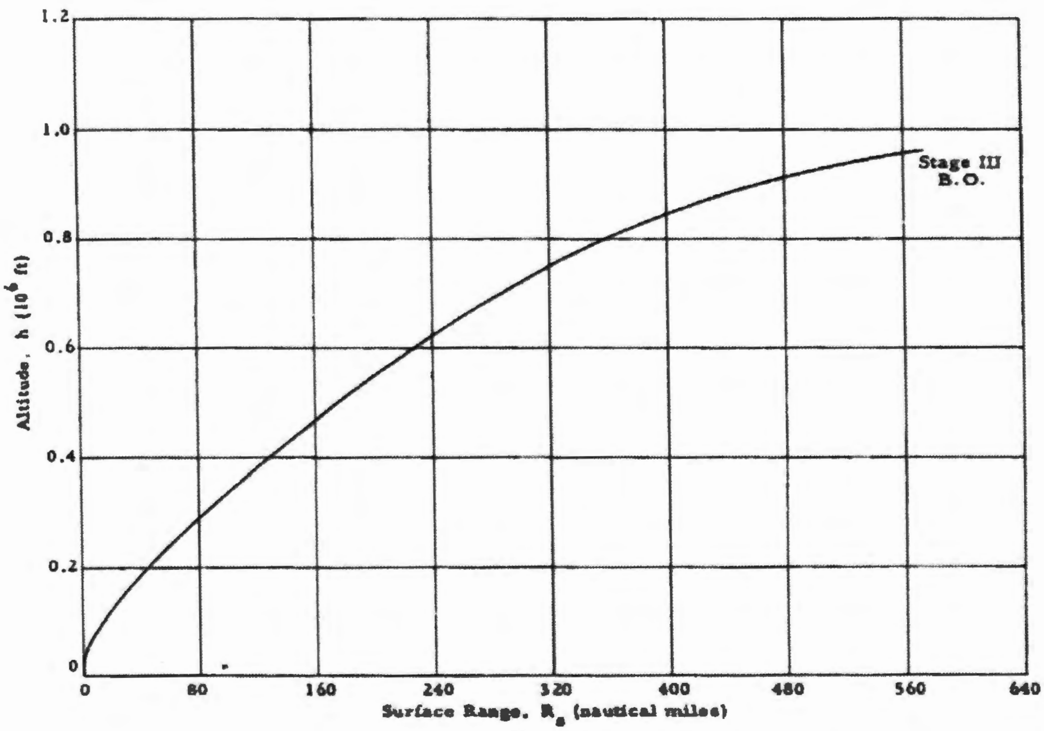


Figure 10. Altitude Versus Surface Range During Powered Flight.
(1001)

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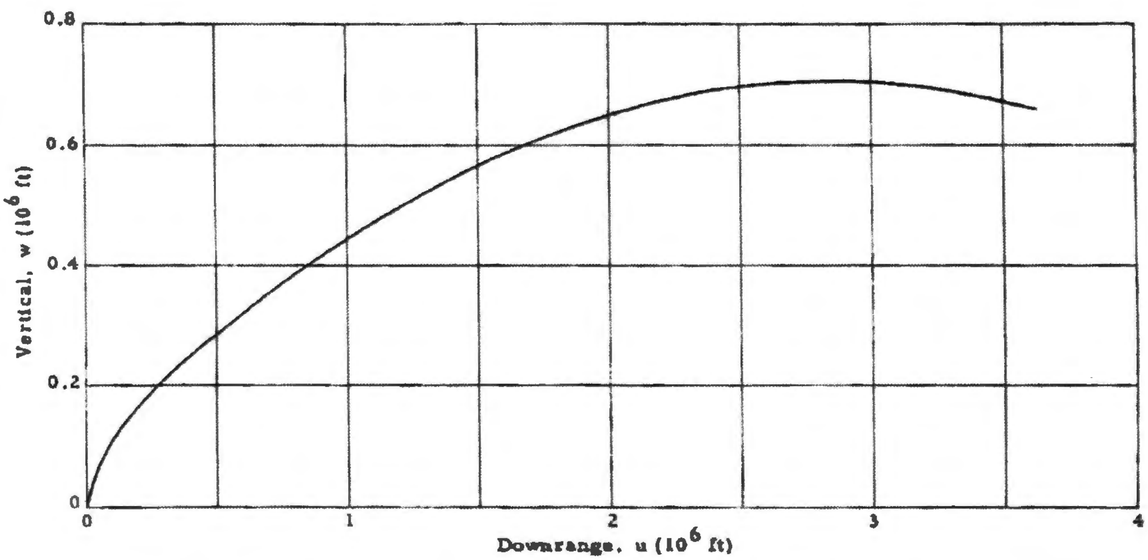

Figure 11. Vertical Distance Versus Downrange Distance During Powered Flight.

(1002)

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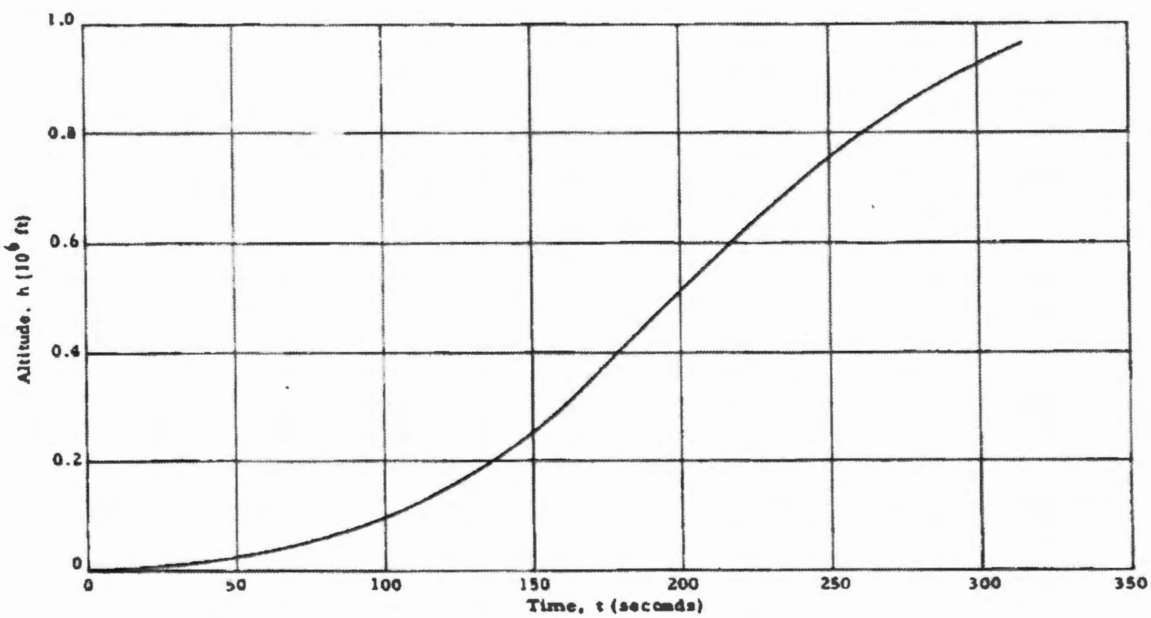

Figure 12. Altitude Versus Time During Powered Flight.

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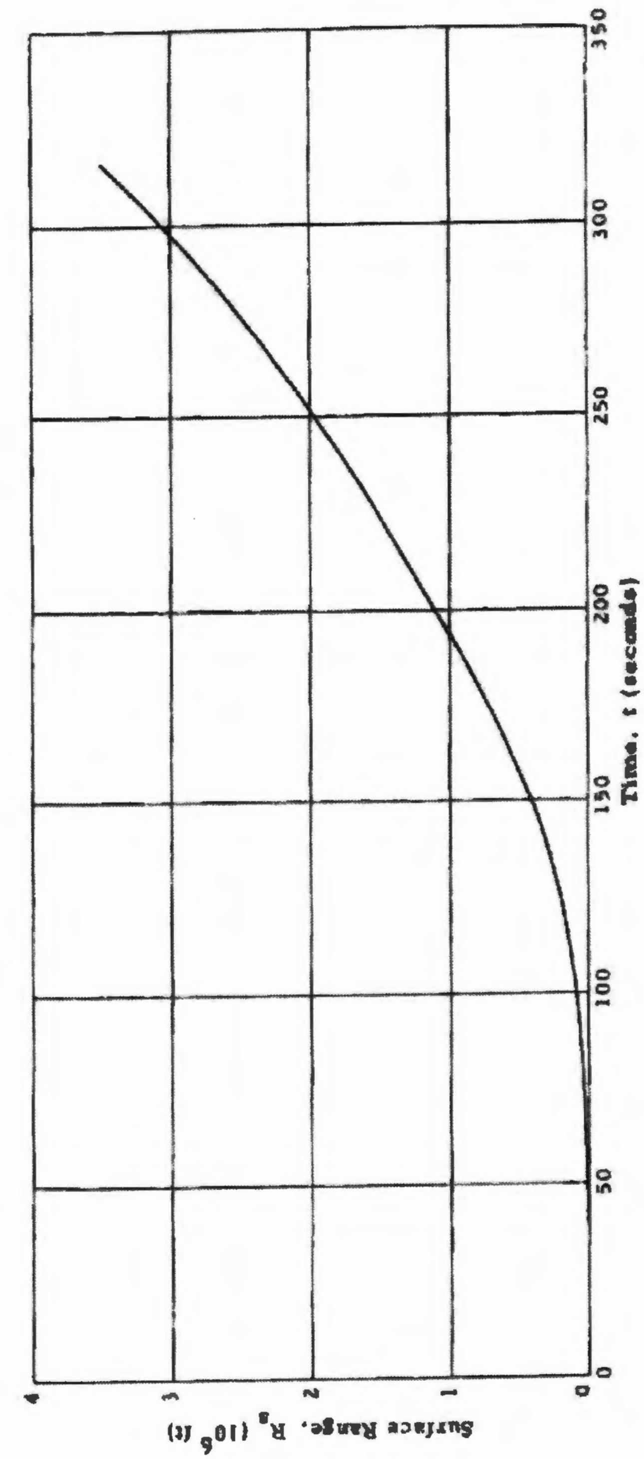

Figure 13. Surface Range During Powered Flight.

1004

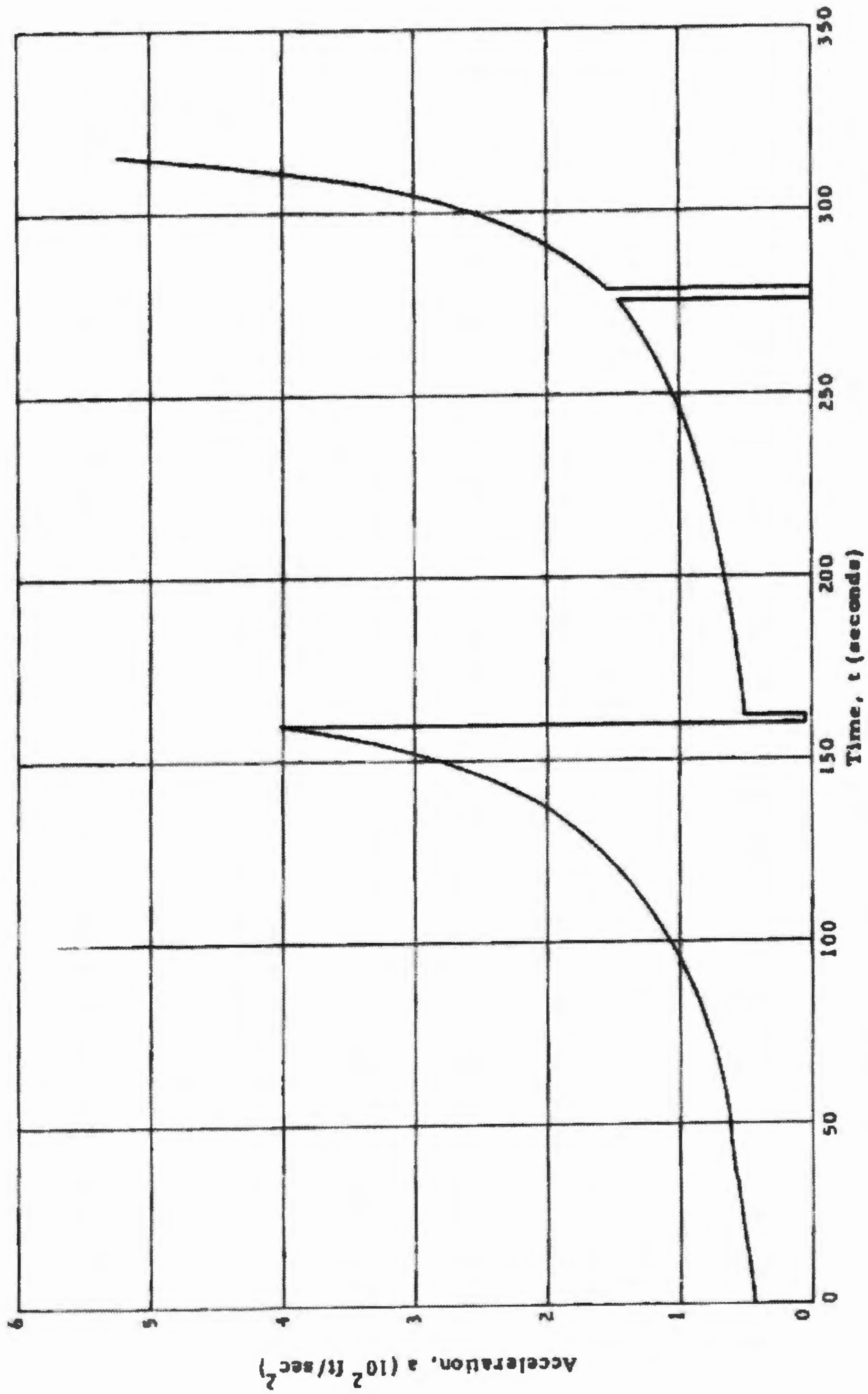

Figure 14. Net Axial Acceleration During Powered Flight.

(1005)

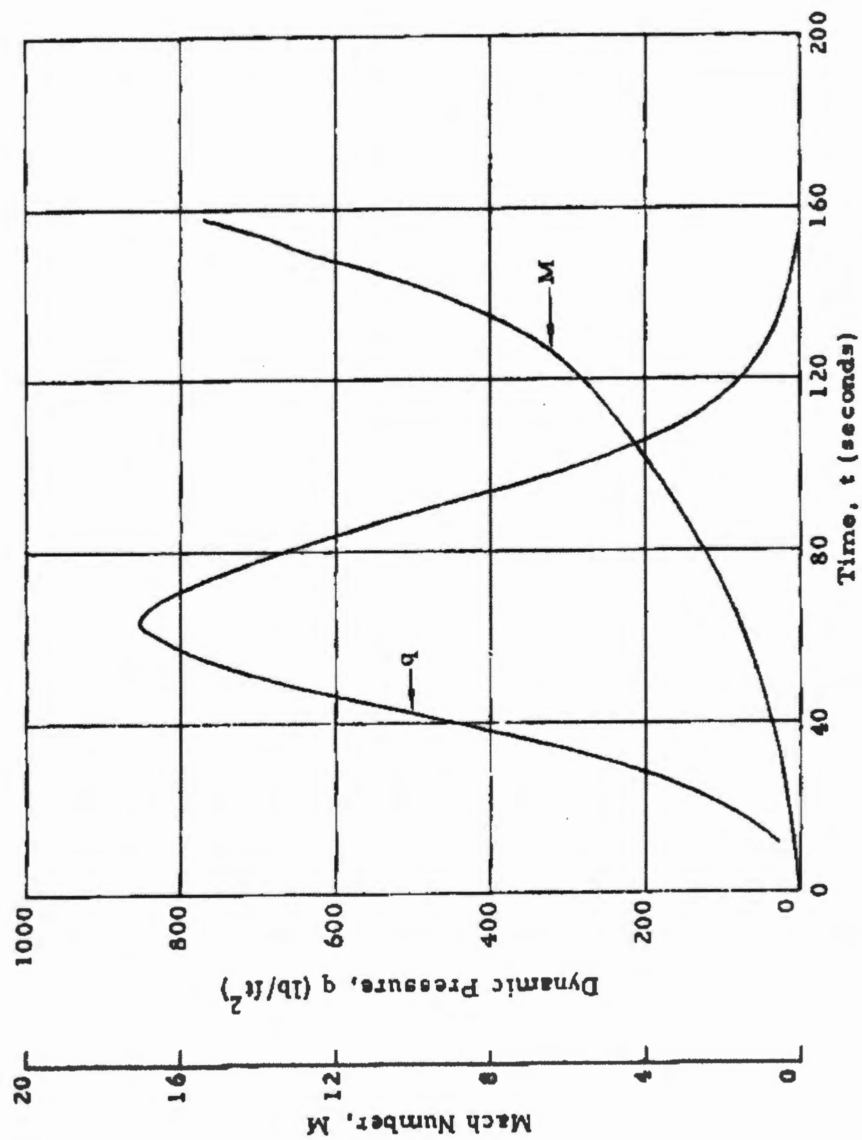

Figure 15. Dynamic Pressure and Mach Number.

(1006)

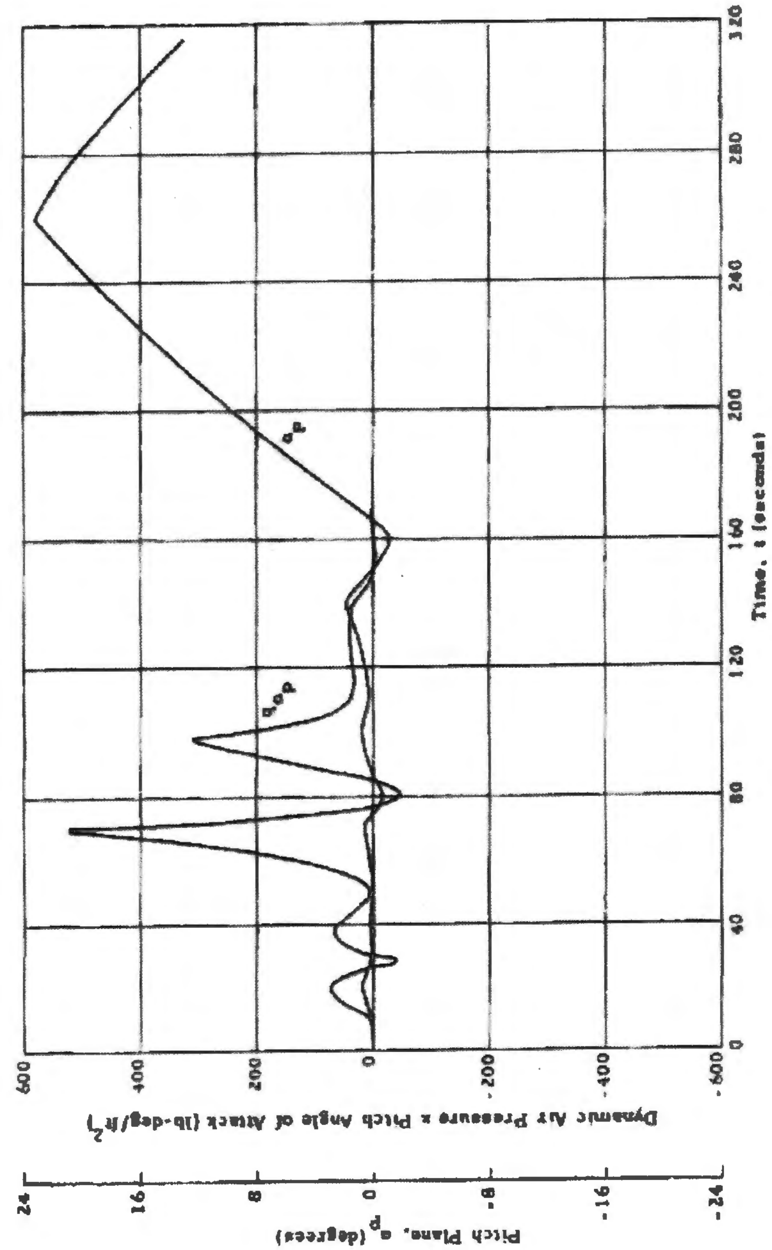

Figure 1[illegible]. α_p and $q\alpha_p$ During Powered Flight.

1011.

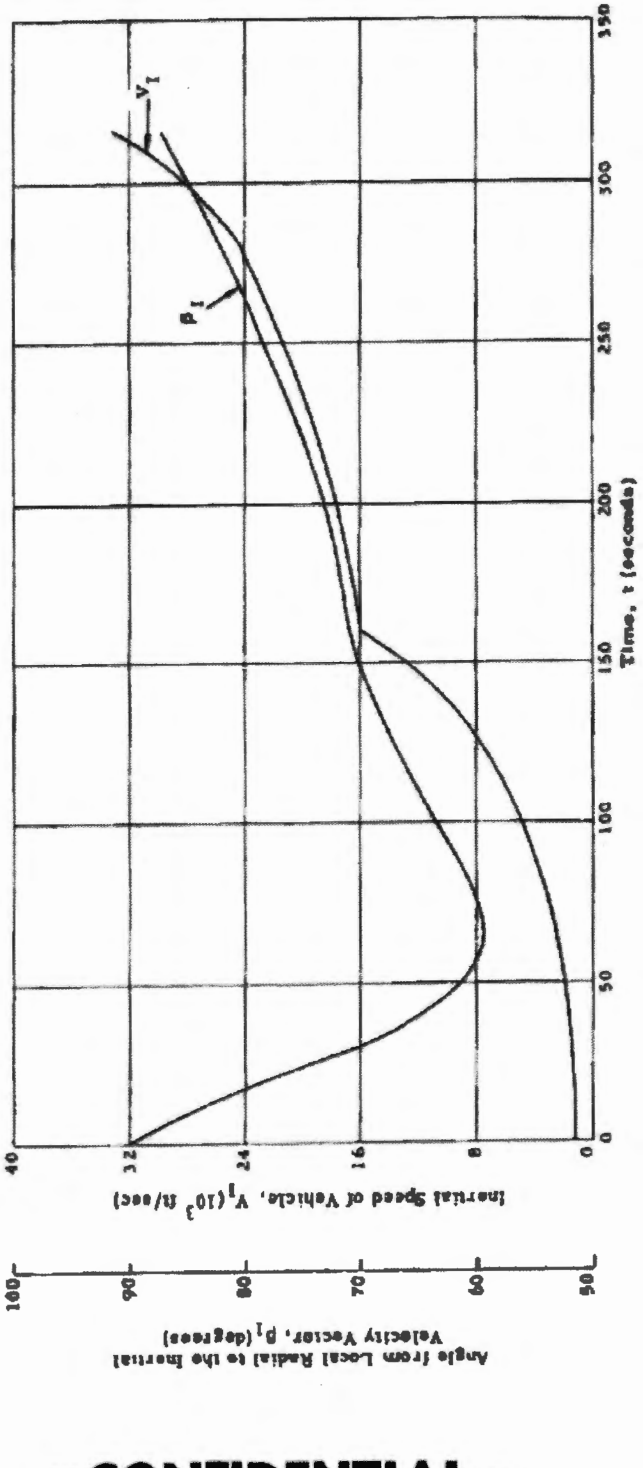

Figure 17. Direction and Magnitude of Velocity Vector.

(1014)

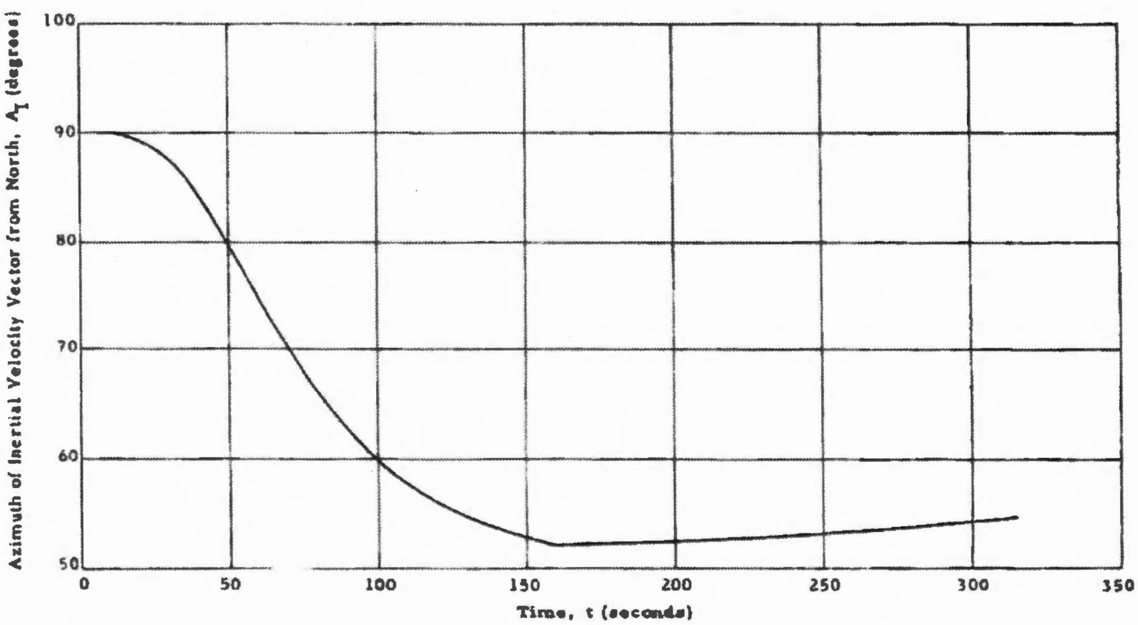

Figure 18. Azimuth of Inertial Velocity Vector Projected in Surface of Earth.

'1000.

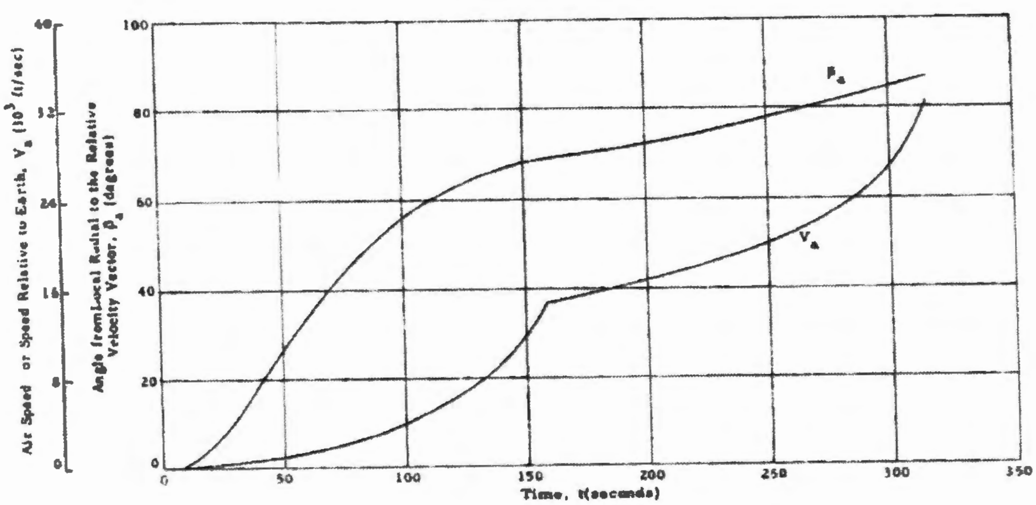

Figure 19. Direction and Magnitude of Air Speed Vector.

.1016)

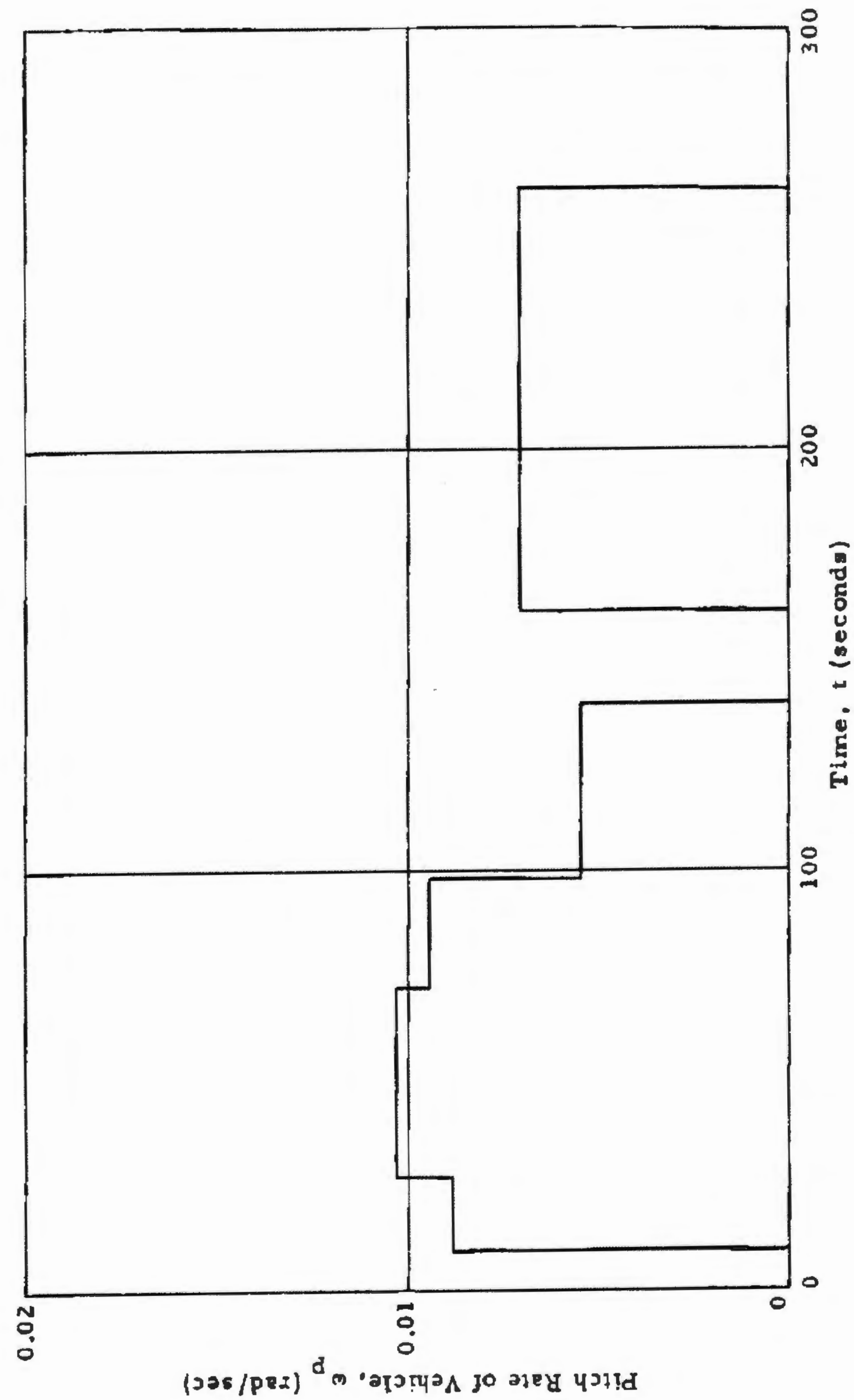

Figure 20. Pitch Rate of Vehicle, ω_p, Versus Time.
(1008)

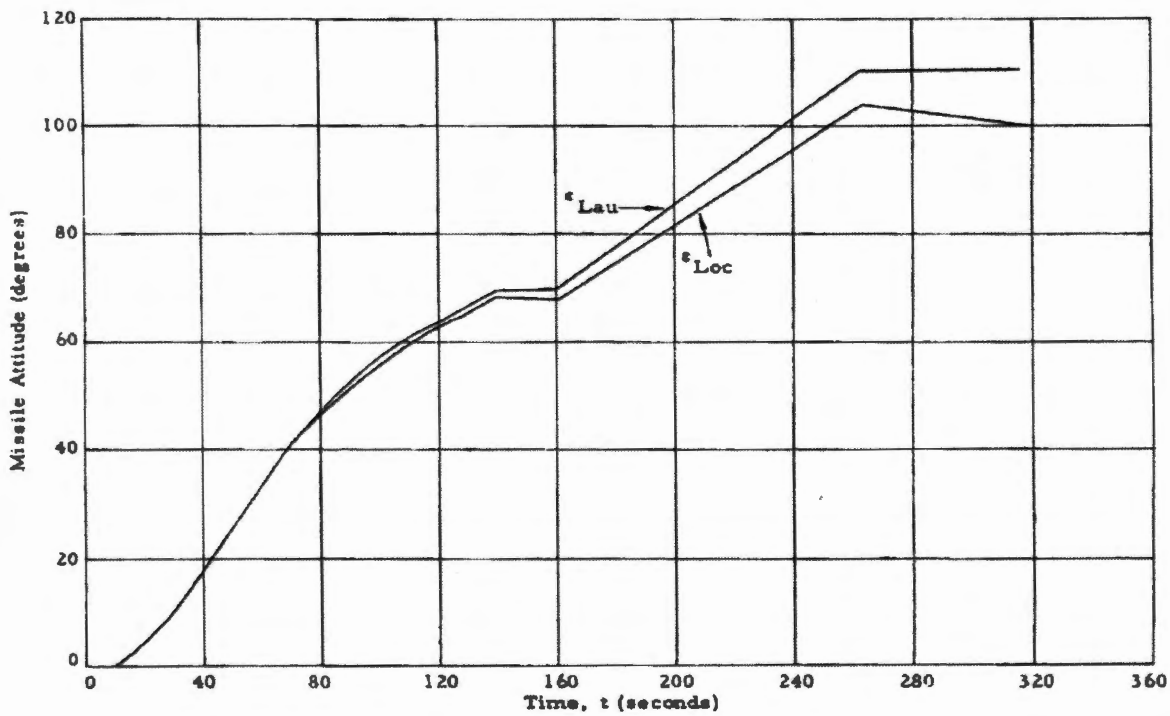

Figure 21. Missile Attitude from Launch Vertical and from Local Vertical.

(1009)

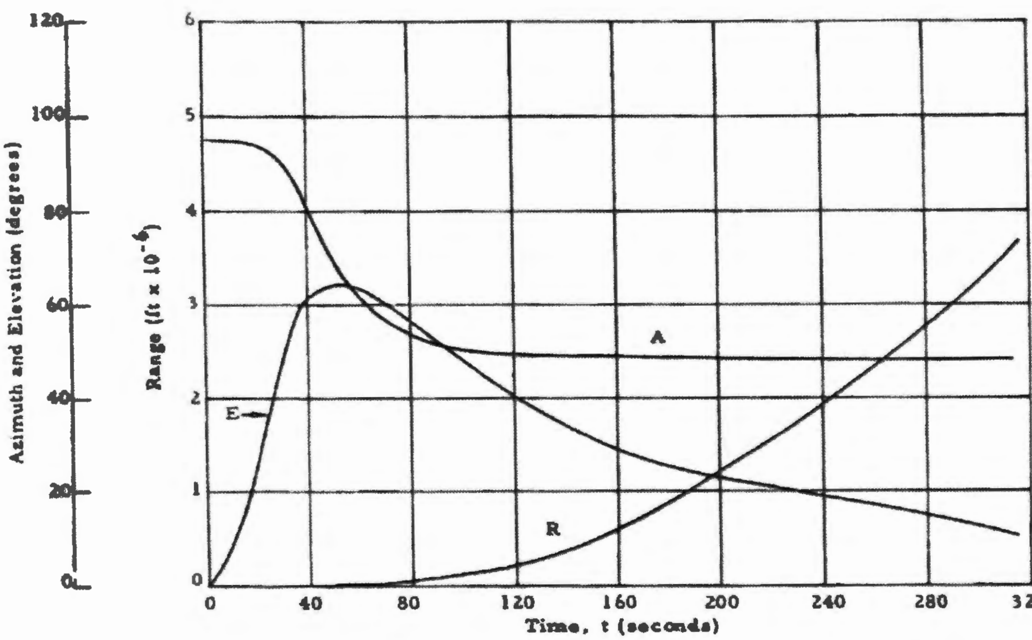

Figure 22. Radar Range, Azimuth, and Elevation During Powered Flight.

(1012)

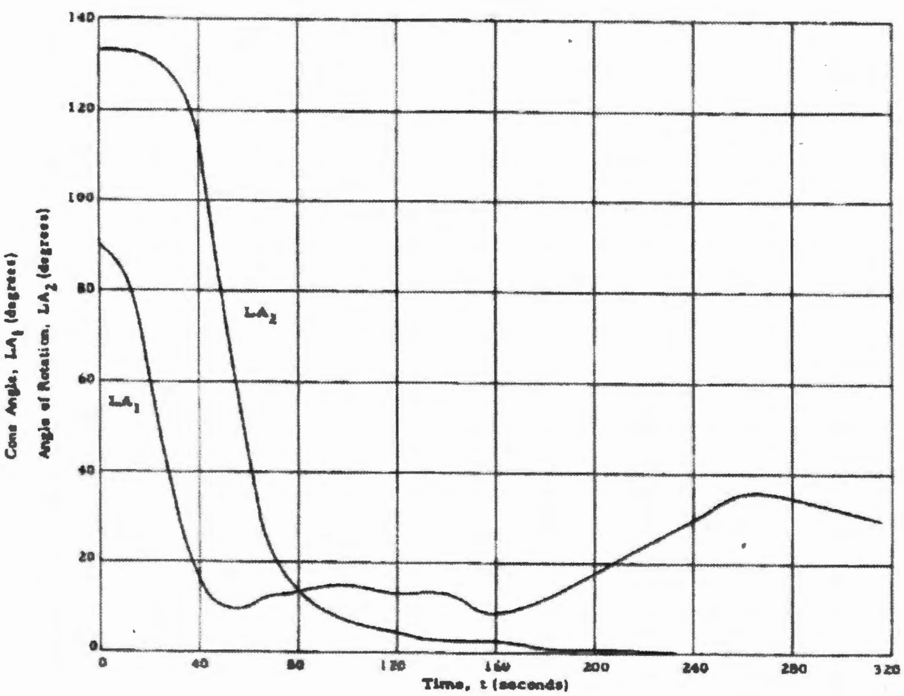

Figure 23. Radar Look Angles During Powered Flight.

1015)

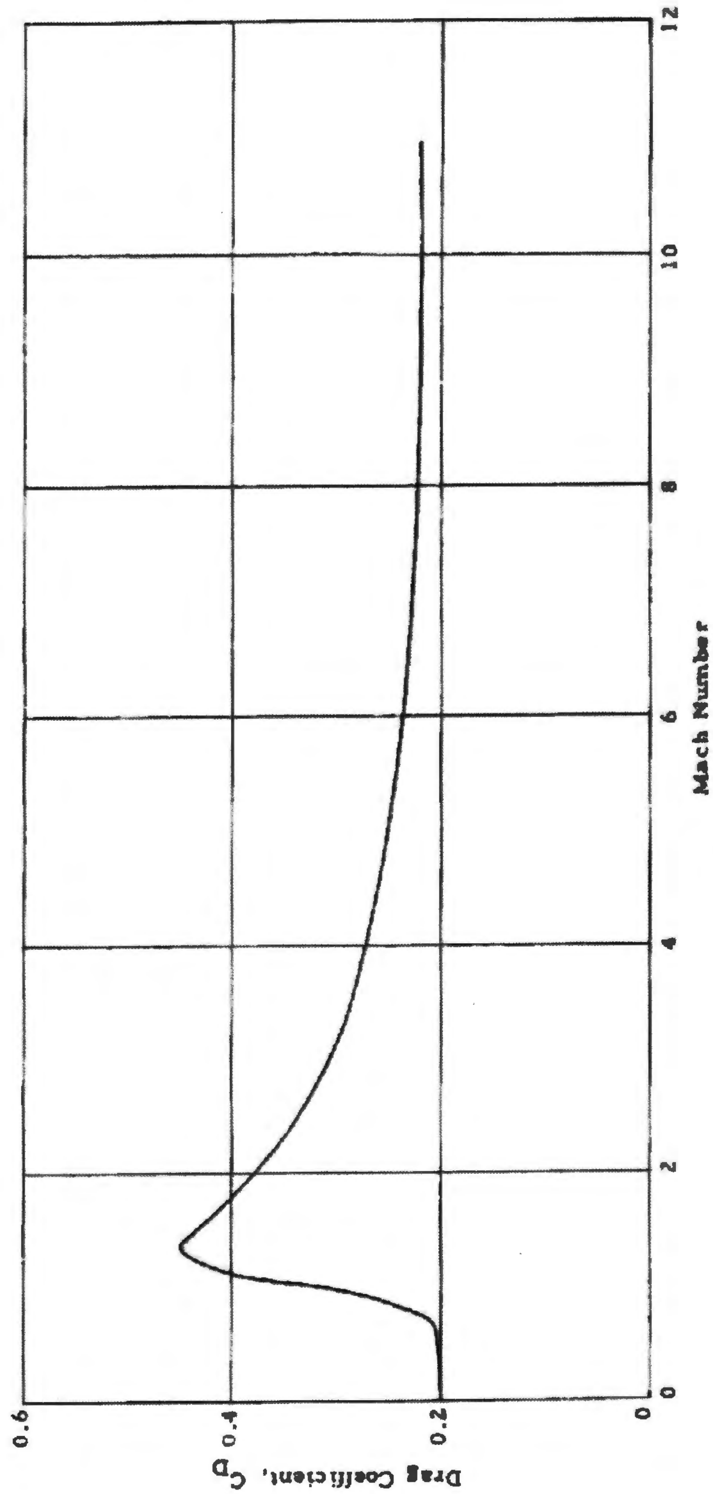

Figure 24. Drag Coefficient Versus Mach Number.

10 4

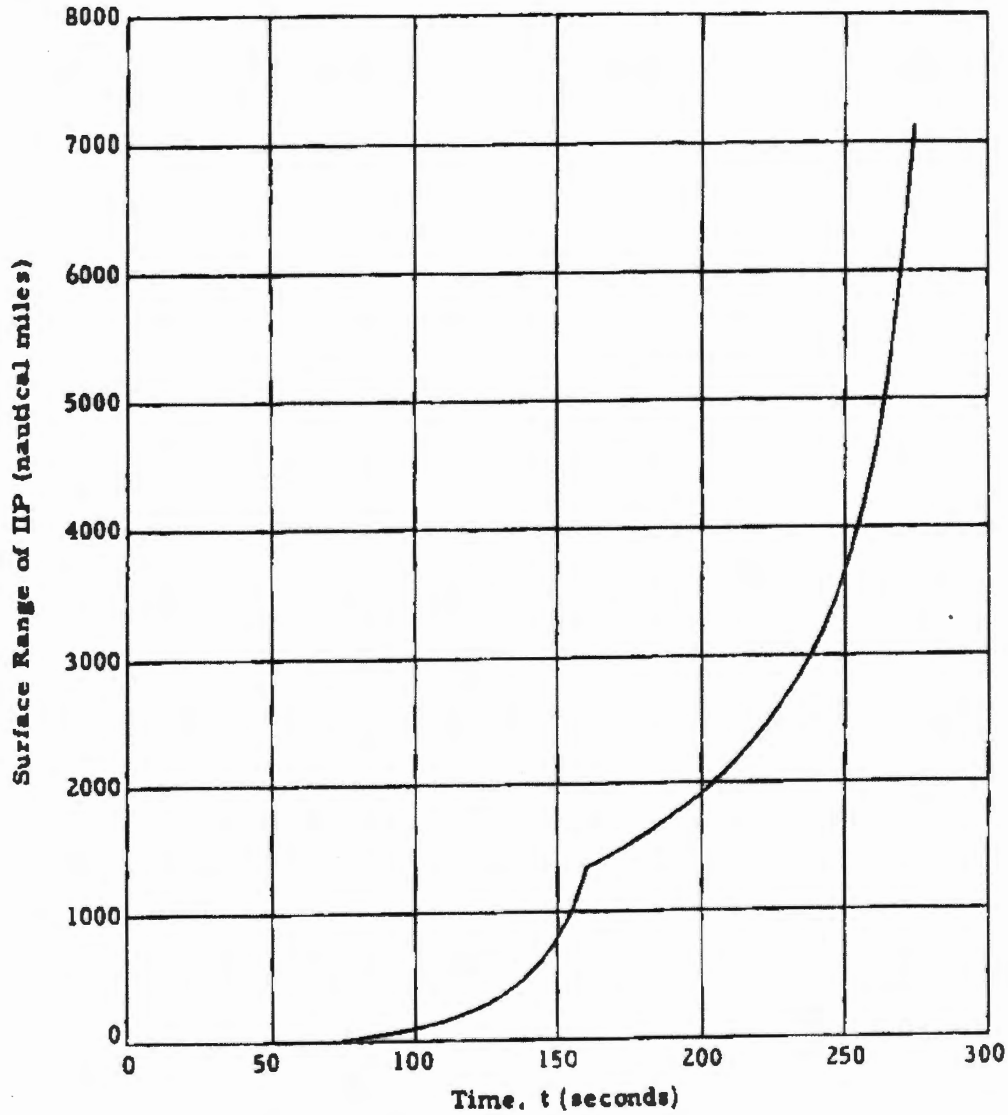

Figure 25. Surface Range of IIP.

(1073)

Table 5. Table of IIP Parameters.

[redacted]

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cutoff at 8-second intervals. The latitude and longitude of IIP is shown, as well as the u, v coordinates, and the downrange and crossrange position on the earth's surface.

Figures 26 through 30 show the projection of the nominal vehicle on the earth for each of the first four days of orbit. Positions of apogee and perigee are indicated. Figures 31 and 32 are projections of the nominal orbit in the xz and xy planes from the first apogee to the second apogee.

Figure 33 shows radar coordinated from Manchester during the first pass indicating when fourth-stage ignition signals can be sent.

Figure 34 is a chart showing the periods of visibility for the first four days from the ground stations at Cape Canaveral, Hawaii, Manchester, Singapore, and Millstone Hill.

2. Assumptions

(a) Rotating, oblate earth

(b) Shape of the earth is given by the AMS ellipsoid

(c) The gravity potential function expressed in Cartesian coordinates is

$$u(x, y, z) = \frac{GM}{a}\left[\frac{a}{r} + \frac{Ja^3}{r^3}\left(\frac{1}{3} - \frac{a^2}{r^2}\right) + \frac{D}{35}\left(35\frac{a^5z^4}{r^9} - 30\frac{a^5z^4}{r^9} - 30\frac{a^5z^2}{r^7} + 3\frac{a^5}{r^5}\right)\right]$$

where $r^2 = x^2 + y^2 + z^2$

$$\frac{GM}{a} = 6.72685 \times 10^8 \ ft^2/sec^2$$

$$J = 0.001638$$

$$D = 0.0000107$$

$$a = 2.092601 \times 10^7 \ ft$$

(d) Minzner revision of ARDA Model Atmosphere, 1956

(e) Mean radius of the earth used in computing surface range of angular range

$$r = 3437.746 \text{ naut mi}$$

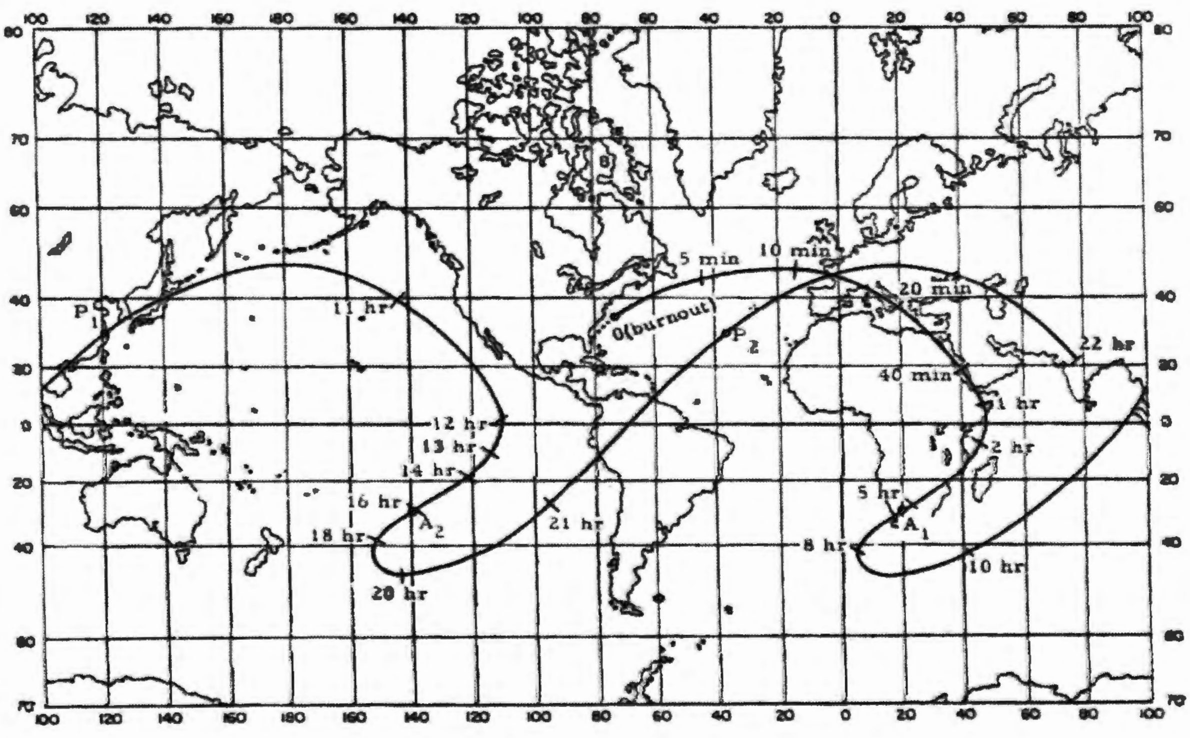

Figure 26. Projection of Nominal Vehicle Orbit; Burnout to 22 Hours. (A_1, First Apogee; A_2, Second; P_1, First Perigee, etc.)

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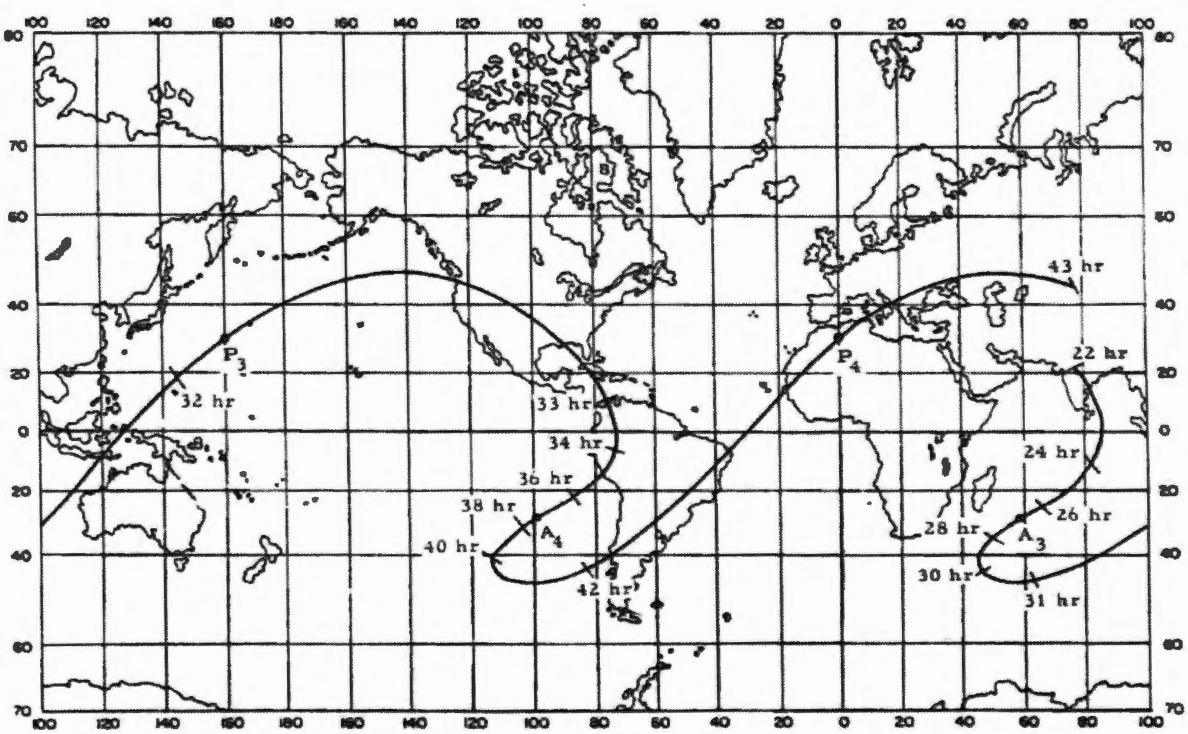

Figure 27. Projection of Nominal Vehicle Orbit; 22 Hours to 43 Hours.

(1033)

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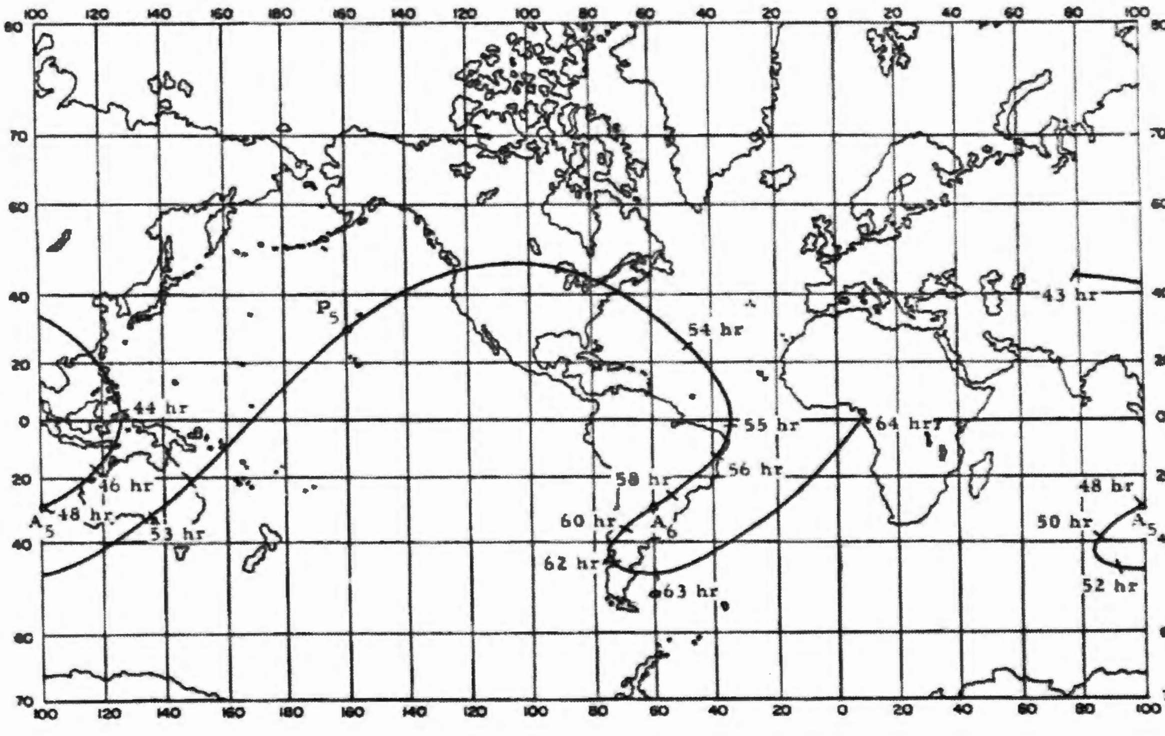

Figure 28. Projection of Nominal Vehicle Orbit; 43 Hours to 64 Hours.

(103a)

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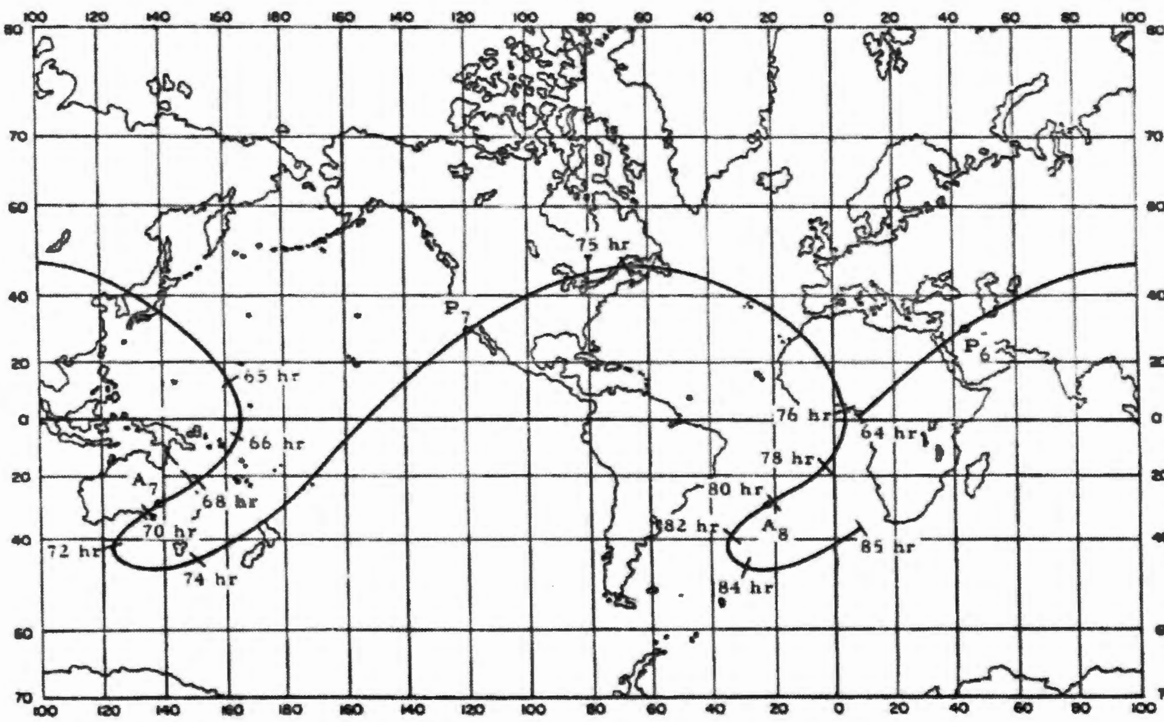

Figure 29. Projection of Nominal Vehicle Orbit; 64 Hours to 85 Hours.

(1035)

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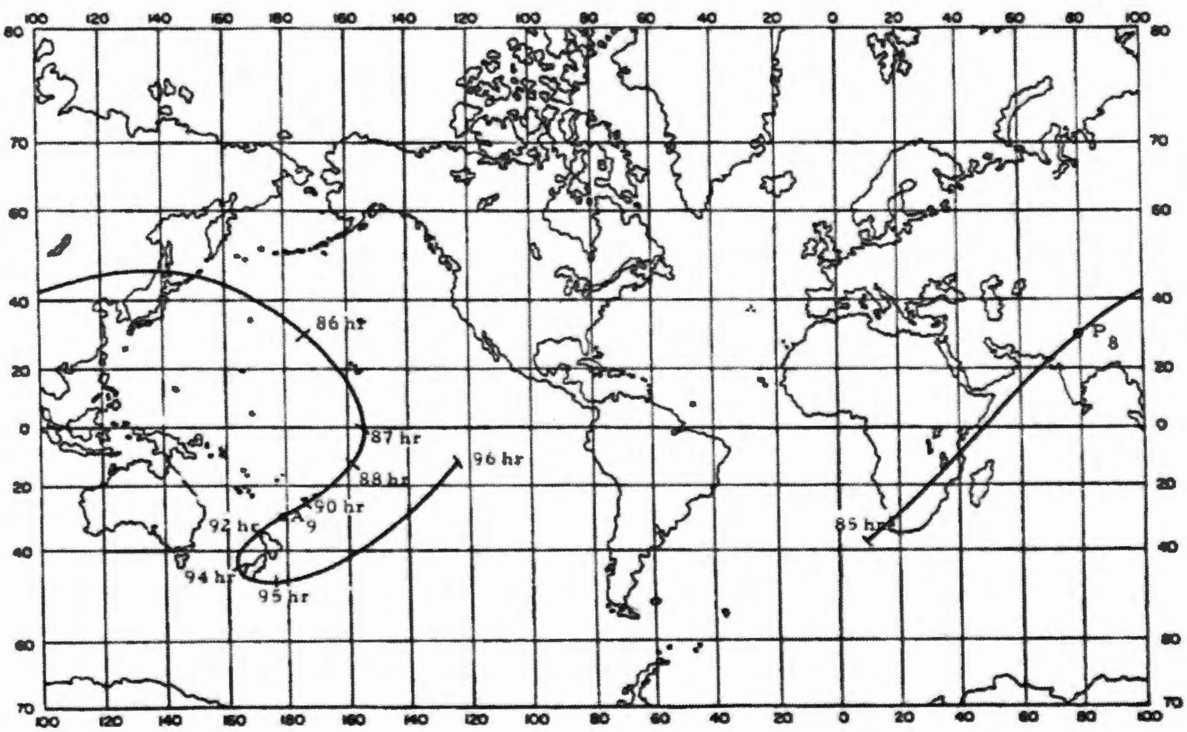

Figure 30. Projection of Nominal Vehicle Orbit; 85 Hours to 96 Hours.

(1036)

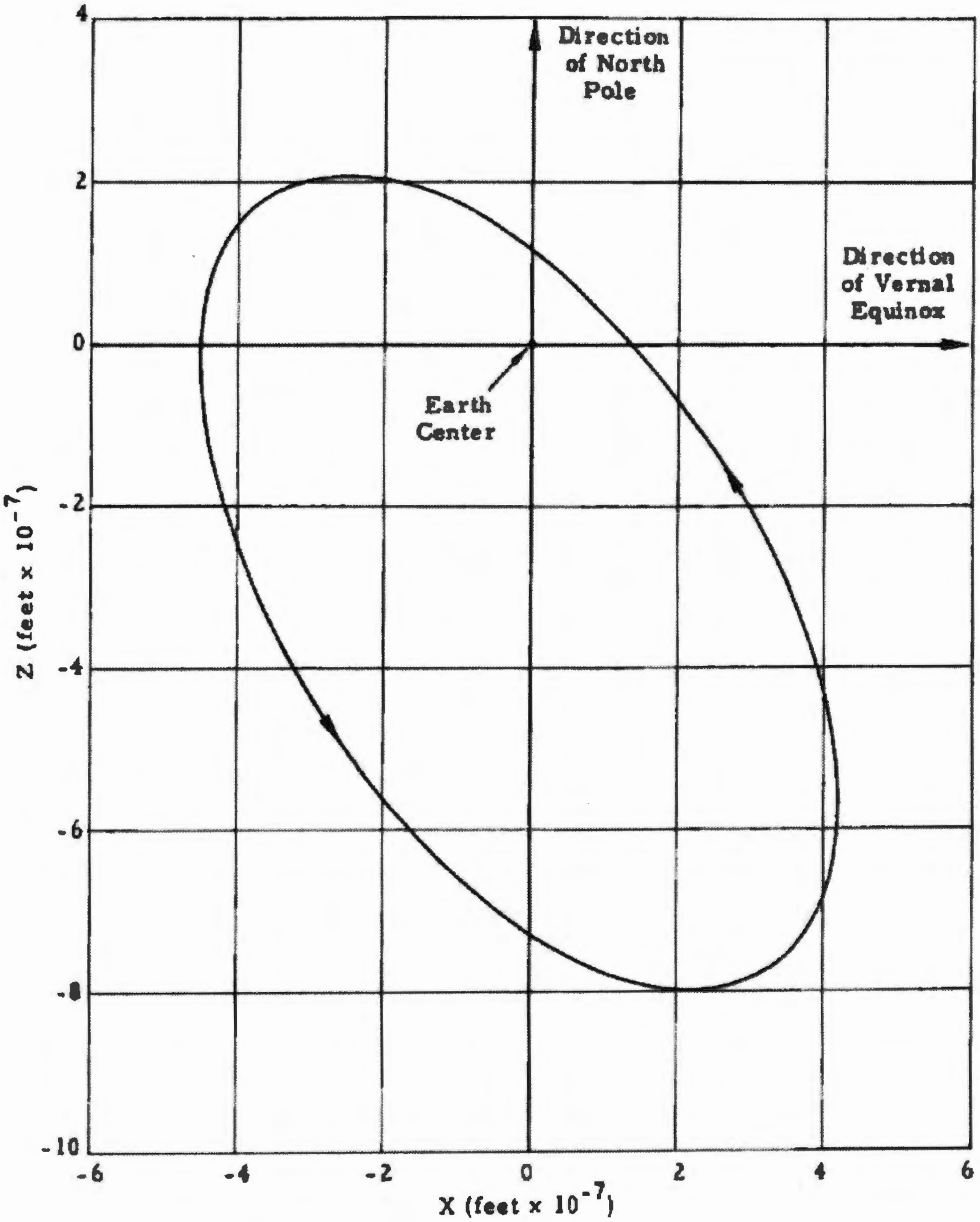

Figure 31. XZ Plane Projection of Nominal Free Flight Trajectory From First Apogee to Second Apogee (In Equatorial Earth Centered Coordinates, X in Direction of Vernal Equinox, Z in Direction of North Pole).

(1807)

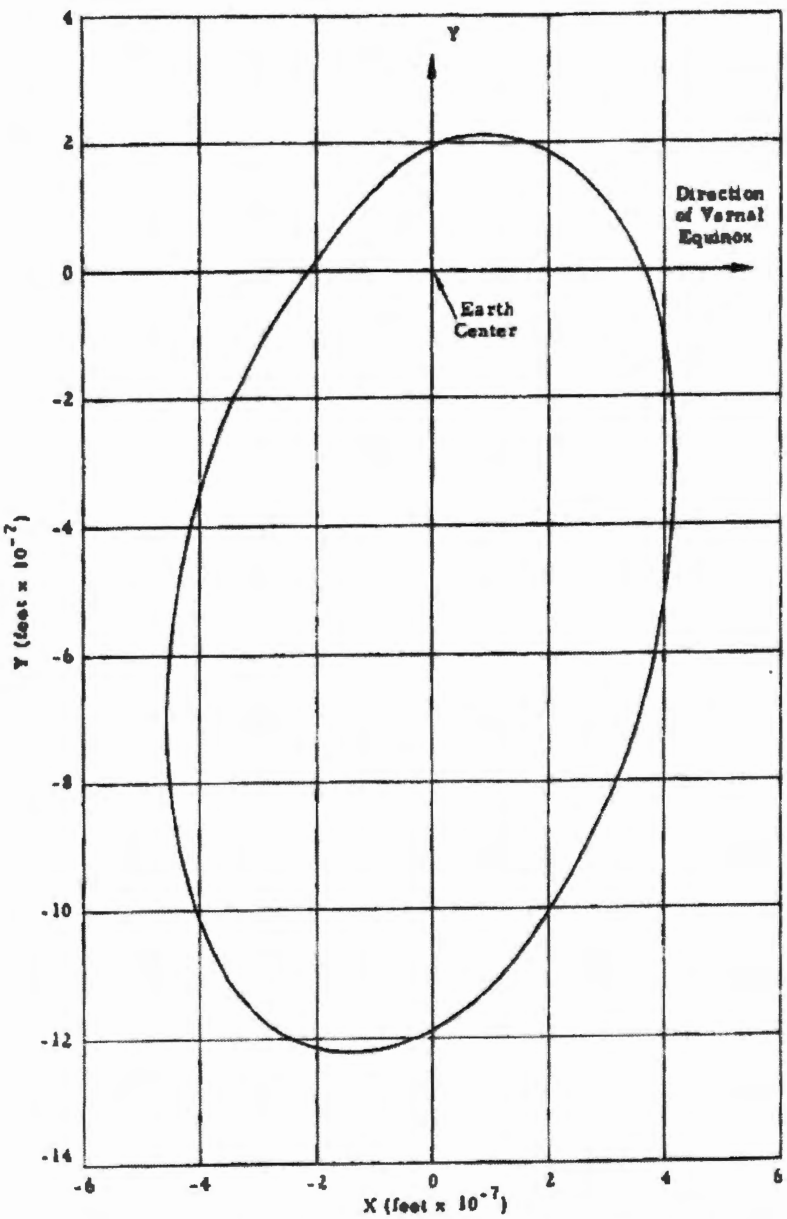

Figure 32. XY Plane Projection of Nominal Free Flight Trajectory from First Apogee to Second Apogee (XY-Plane is Equatorial Plane).

1010

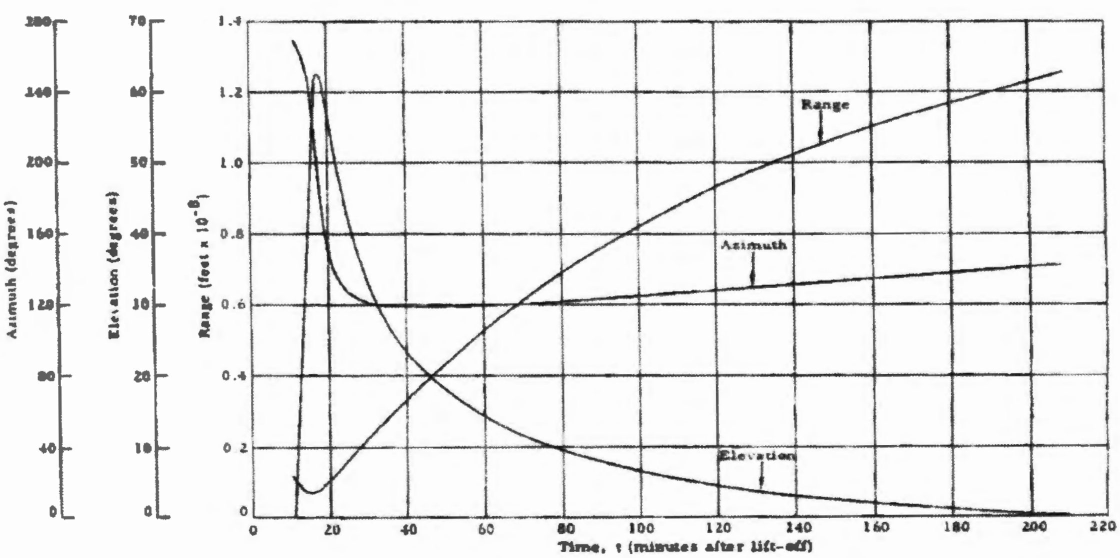

Figure 33. Radar Coordinants from Manchester During Free Flight (First Pass).

(1013)

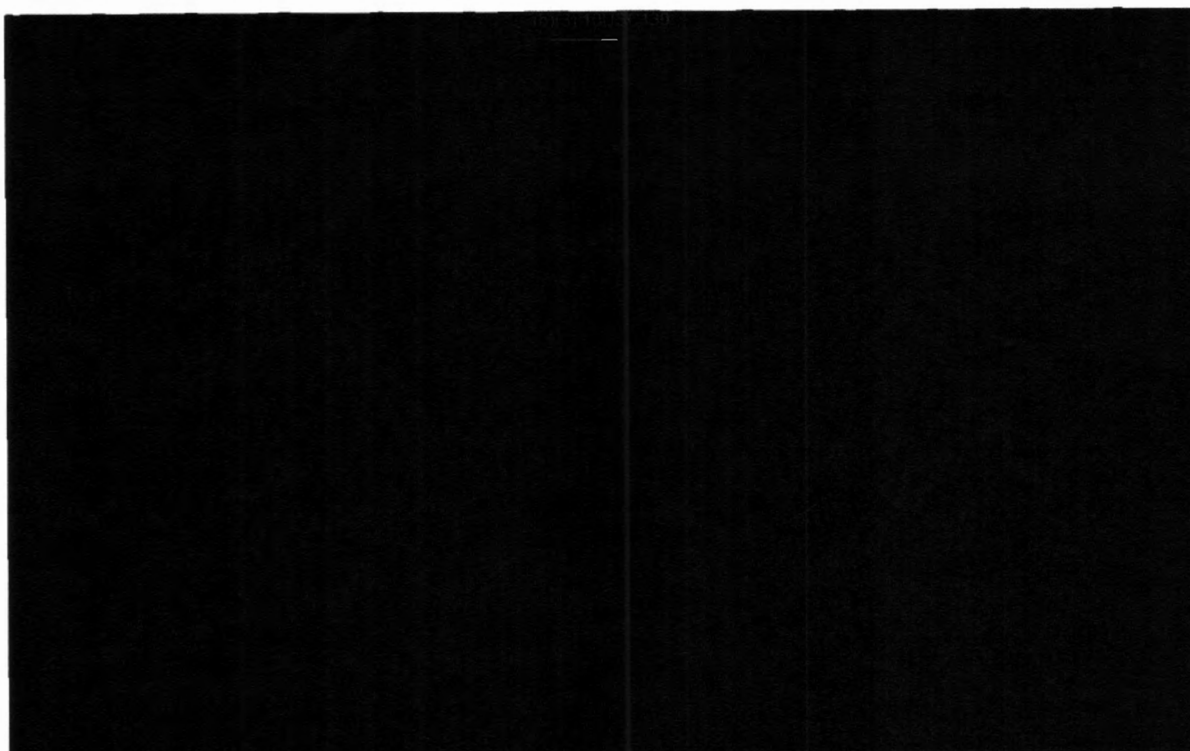

Figure 34. Periods of Visibility from Tracking Stations for First Four Days.

(1037)

(f) One nautical mile = 6076.1033 ft

(g) Weight-to-mass conversion factor = 32.174 ft/sec^2

3. Definitions of Symbols

t	-	time from lift-off (sec)
x, y, z	-	inertial coordinates of the vehicle with the origin at the center of the earth. The xz plane contains the initial position of the launch point, z is along the north pole (ft)
$\dot{x}, \dot{y}, \dot{z}$	-	the components of velocity in the x, y, z system (ft/sec)
u, v, w	-	earth-fixed coordinates of the vehicle with the origin at the launch pad, the positive u axis directed downrange along the launch azimuth, the positive v axis directed to the left of an observer looking downrange, and the positive w axis upward (ft)
$\dot{u}, \dot{v}, \dot{w}$	-	the components of velocity in the u, v, w system (ft/sec)
r	-	radial distance from vehicle to the center of the earth (ft)
θ	-	longitude of vehicle measured from launch pad (deg)
ϕ	-	geocentric latitude of vehicle (deg)
$\dot{R}_1$	-	radar range rate from doppler site at Cape Vanaveral (ft/sec)
LA_1, LA_2	-	radar look angles from doppler site (deg) LA_1 is angle between vehicle roll axis and radar line of sight LA_2 is angle between vehicle yaw axis and radar line of sight projected onto the pitch-yaw plane
V_I	-	inertial speed of vehicle (ft/sec)
β_I	-	angle from local radial to the inertial velocity vector (deg)
V_a	-	air speed, or speed relative to earth (ft/sec)
β_a	-	angle from local radial to the relative velocity vector (deg)
V_ξ	-	integrated axial acceleration (ft/sec)

h	-	altitude of vehicle above reference ellipsoid, measured along local radial (ft)
F	-	total thrust (lb)
a	-	effective acceleration, $(F - D)/M$ (ft/sec^2)
W_m	-	vehicle weight (lb)
$\dot{W}_m$	-	vehicle weight rate (lb/sec)
α_p, α_y	-	pitch and yaw plane components of angle of attack (deg)
R_s	-	downrange distance measured along surface of spherical earth (ft)
θ_t	-	longitude of vehicle measured from initial position of launch pad (deg)
ω_p	-	pitch rate of vehicle (rad/sec)
ϵ_{Lau}	-	body attitude measured from initial launch radial (deg)
ϵ_{Loc}	-	body attitude measured from local radial (deg)
A_I	-	azimuth of inertial velocity vector from north (deg)
D	-	aerodynamic drag (lb)
C_D	-	zero lift aerodynamic drag coefficient
N	-	aerodynamic normal force (lb)
q	-	dynamic air pressure (lb/ft^2)
M	-	Mach number
$q\alpha_p$	-	dynamic air pressure x pitch angle of attack (lb-deg/ft^2)
R_1, E_1, A_1	-	radar range, elevation, and aximuth from Cape Canaveral doppler site
R_2, E_2, A_2	-	radar range, elevation, and aximuth from Millstone radar

POWERED FLIGHT PRINTOUT KEY

1	x	y	z	$\dot{x}$	$\dot{y}$	$\dot{z}$
2	u	v	w	$\dot{u}$	$\dot{v}$	$\dot{w}$
3	r	θ	ϕ	$\dot{R}_1$	LA_1	LA_2
4	V_1	β_I	V_a	β_a	V_ξ	h
5	F	a	W_m	$\dot{W}_m$	α_p	α_y
6	R_s	θ_t	ω_p	ϵ_{Lau}	ϵ_{Loc}	A_I
7	D	C_D	N	q	M	$q\alpha_p$
8	R_1	E_1	A_1	R_2	E_2	A_2

	I-D [illegible] STUDIES		OSWALD A-3 POWER		M. WOODFIELD	
	45.000	.00000	-3000	.00706	2.00	[illegible]
	112257	14149.00000	.00000000	50.3	PFT1-2	[illegible]
	14149	2.00000	.00000000	50.3	2000	[illegible]
	-.157	1515.[illegible]	.00175356	5.6	PFT2-1	PFF2-1
	25.[illegible]	[illegible]	1515.[illegible]			
	650.7	195.00000	.00000000	5.6		
	28.44687	28.44815	-.01388	42.81760		
.02000	.18413437 08	.[illegible]-39	.99785185 07	.00000000-39	.1342[illegible] 04	.00000000-39
	.15356025 03	.17354592 00	.25000000 00	.11339490-04	-.10210123-04	.00000000-39
	.20910126 28	.00000000-39	.28285294 02	.00000000-39	.99992788 02	.13299772 03
	.13427281 04	.89999999 02	.00000000-39	.00000000-39	.00000000-39	.00000000-39
	.15324800 06	.43922[illegible] 02	.11225700 06	-.61619999 03	[illegible]	[illegible]
	.00000000-39	.00000000-39	.[illegible]-39	.00000000-39	.16131967 00	.20000003 02
	.00000000-39	.20400000 00	.00000000-39	.00000000-39	.00000000-39	.00000000-39
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2.00000	.18413459 08	.26556534 04	.69785900 07	.20095779 02	[illegible]	[illegible]
	.23260990 00	.16733888 00	.24000000 07	-.66607975-02	[illegible]	[illegible]
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	.13439371 04	.45971667 [illegible]	.24101655 02	.16253776 00	.88253436 02	.23750000 02
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	.45288700-01	.77705773 02	.22175000 03	-.93338000-01	[illegible]	[illegible]
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